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WIND-TUNNEL PROCEDURE FOR DETERMINATION OF CRITICAL STABILITY AND CONTROL CHARACTERISTICS OF AIRPLANES

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WASHINGTON

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SUMMARY

This report outlines the flight conditions that are usually critical in determining the design of components of an airplane which affect its stability and control characteristics. The wind-tunnel tests necessary to determine the pertinent data for these conditions are indicated, and the methods of computation used to translate these data into characteristics which define the flying qualities of the airplane are illustrated.

INTRODUCTION

The development of flying-qualities specifications (references 1, 2, and 3) has established specific criteria with which the characteristics of an airplane normally will be compared. The problems posed in the preliminary design of an airplane is the determination of which of these criteria will influence the design of the various components of the airplane that affect the stability and control characteristics, and the magnitude of the effect. As an aid in this design problem, methods have been developed by which the data, obtained from wind-tunnel tests of powered models, can be translated into flying-qualities characteristics observable in flight tests (in the terms in which the flying-qualities specifications are written). Application of these methods to six different airplanes has indicated that the same requirements represent the critical conditions on all conventional airplanes, and that if these conditions are met,

it will follow that the remainder of the specifications will be satisfied. By permitting concentration on these few conditions, a considerable simplification of the design process results.

It is the purpose of this report to outline the critical conditions for each component of the airplane, to indicate the wind-tunnel tests necessary to determine the pertinent data, and to illustrate the methods of computation used to translate these data into characteristics which define the flying qualities of an airplane.

DISCUSSION

The flying-qualities requirements can be stated under three major headings:

1. Stability shall exist under specified conditions.

2. Control shall exist under specified conditions.

3. Control forces shall be kept within specified limits.

Each of these requirements is, to some extent, contradictory to the other two and, furthermore, airplanes now have been developed to such sizes and powers that the attainment of all three requirements is quite difficult. Hence, despite the fact that from the ultimate flying-qualities standpoint it is desirable to satisfy some of the requirements by as ample a margin as possible, the designer normally will find it expedient to base his original design on small margins, in order to minimize the difficulty of compromising conflicting requirements. If this is not done for one requirement, the attainment of the other two by normal means may be impossible.

To illustrate this point, the horizontal tail on a typical high-powered, single-engine airplane must be the <u>smallest</u> which will give the required stability in a rated-power climb, and the elevator must be the <u>smallest</u> which will give the required control in landing, in order to keep the balance requirements for low control forces in accelerated maneuvers within reasonable limits. With regard to wing dihedral, care must be taken not to exceed the amount required for the maintenance of lateral stability in the low-speed, high-power condition where the dihedral effect will be

minimum, or excessive dihedral effect will result at high speeds. The size of the rudder must be limited to the smallest that will give adequate control in order to keep the rudder-pedal forces within the required limits.

If it is assumed that the preliminary design has been completed on the above basis, it will be the function of the first wind-tunnel tests to obtain data from which any re-adjustments of the airplane components, necessary to secure satisfactory characteristics, can be determined. As conceived herein, the first series of wind-tunnel tests would be restricted to the critical conditions with regard to each characteristic. A series of tests sufficiently complete to form a basis for a more general flying-qualities prediction, or an analysis of secondary effects, would not be made until the changes shown to be necessary by the first series of tests had been incorporated in the model. An outline for such a preliminary series of tests as just discussed is given in tables I, II, and III for a single-engine airplane and in tables I, II, and IV for a twin-engine airplane. An attempt has been made to make these tables self-explanatory when considered in the light of a flying-qualities specification (references 1, 2, and 3). Figures 1 to 16 present a typical set of results. The method of translating the wind-tunnel results into the terms of the flying-qualities specification is outlined on these figures.

The choice of critical conditions and the tables have been made after a detailed study of the characteristics of 3 typical single-engine airplanes and 3 twin-engine airplanes <u>with right-hand rotating propellers</u>. In each case it was found that if the 10 major points as outlined were satisfied, the other characteristics called for in the flying-qualities specifications would be met. It is believed that this conclusion will be similar for other conventional airplanes.

Each of the 10 items listed in the tables is directed toward 1 major variable in the airplane design. Thus, in the usual case

<u>Horizontal tail size</u> will be determined by item I.

<u>Elevator size</u> will be determined by item II.

<u>Elevator balance</u> will be determined by item III.

<u>Minimum dihedral</u> will be determined by item IV.

Maximum dihedral will be determined by item V.

Aileron size will be determined by item VI.

Aileron balance will be determined by item VII.

Vertical tail size will be determined by item VIII.

Rudder size will be determined by item IX.

Rudder balance will be determined by Item X.

Obviously there is a closer interrelation among the characteristics than the above listing implies, and important changes can be required after consideration of "secondary" variables. However, to a first approximation the variables listed will establish the airplane stability and control characteristics after the first basic arrangement of wing and fuselage is established. Changes in other features of the airplane components will normally be in the nature of refinements, rather than major changes.

The surface deflections given in the text are only representative values corresponding to the range of deflections needed in ascertaining the flying qualities of the airplanes upon which the study has been based. An optimum selection can be best determined from a cursory examination of the basic runs with control surfaces neutral, with due regard for the maximum deflections upon which the design is based. It will be noted that tail-off runs are called for in the tables only when they are necessary for the computation of the flying qualities. However, in order to provide data, which will aid in any necessary redesign, the addition of a tail-off run for other test conditions is considered desirable.

A typical set of data as obtained from the runs called for on the tables is shown in the figures, and the cross plots and computation methods necessary to reduce these data to the form of the flying-qualities characteristics are outlined. As in the table, these figures are intended to be in such detail as to require no further explanation. In the computation procedure certain simplifications and assumptions have been made, but it is believed that all factors which will bear an important influence on the final result have been included.

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APPENDIX

SYMBOLS

δ deflection of control surface

$\Delta\alpha$ change in angle of attack at wing due to ground effect or change in angle of attack (over aileron station) due to roll

$\Delta\epsilon_1$ change in downwash at tail due to ground effect

$\Delta\alpha_{t_1}$ change in angle of attack of tail due to ground effect

$\Delta\alpha_{t_2}$ change in angle of attack of tail due to induced angle in accelerated flight

C_{L_W} lift coefficient of wing and fuselage (exclusive of tail)

i_t angle of incidence of tail

C_h hinge-moment coefficient

ψ angle of yaw

β angle of sideslip

T_c propeller thrust coefficient $= \dfrac{\text{Thrust}}{\rho V^2 D^2}$

C_{n_β} yawing moment due to sideslip

C_{l_β} rolling moment due to sideslip

C_{l_p} rolling moment due to rolling

C_{l_a} rolling moment due to aileron deflection

F stick force, pounds

V_i indicated airspeed

l_H length from center of gravity to 25 percent M.A.C. of horizontal tail

n normal acceleration

g acceleration due to gravity

p rolling velocity, radians per second

b wing span, feet

Subscripts

e elevator

r rudder

a_L left aileron

a_R right aileron

t tail

NOTE: Stability axes have been used in the presentation of the data.

Positive deflection of control surface is in the direction which will produce a positive force (not necessarily a positive moment).

REFERENCES

1. Gilruth, R. R.: Requirements for Satisfactory Flying Qualities of Airplanes. NACA ACR, April 1941. (Classification changed to "restricted" Oct. 1943).

2. Anon.: Stability and Control Characteristics for Airplanes. Spec. No. C-1815, Army Air Forces, Aug. 31, 1943.

3. Anon.: Specification for Stability and Control Characteristics of Airplanes. Spec. No. SR-119, Bur. of Aero., Navy Dept., Oct. 1, 1942.

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5. Recant, Isidore G.: Wind-Tunnel Investigation of Ground Effect on Wings with Flaps. T.N. No. 705, NACA, May 1939.

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TABLE I

Longitudinal Characteristics
(Single- and Twin-Engine Airplanes)

Item	Purpose and requirement	Critical condition	Run No.	Description of run	Fig. No.	Remarks
I	To determine if the horizontal tail is large enough to meet requirement of stick-fixed stability and if the elevator-floating characteristics are such as to maintain stick-free stability under the specified flight condition.	Critical condition will be in a rated-power climb, where destabilizing effects of power at normal flight speeds will be maximum. Speed range over which stability is required is to be determined from particular specification being followed.		Polars with rated power, flaps and gear up.	1	
			1a	$\delta_e = 0^\circ$		
			1b	$\delta_e = -5^\circ$		
		or				
		Critical condition may occur in the approach, with flaps and gear down and 50-percent normal rated power.		Polars with 50-percent normal rated power, flaps and gear down.	2	
			2a	$\delta_e = 0^\circ$		
			2b	$\delta_e = -5^\circ$		
			2c	$\delta_e = -10^\circ$		
II	To determine if elevator is large enough for necessary control under all normal flight conditions.	Critical condition will be in landing where forwardmost center-of-gravity location and ground effect will require maximum up elevator to secure landing attitude.		Polars with propeller windmilling, flaps and gear down.	3	Runs (4a,b,c) with increased incidence are for the purpose of determining dC_m/di_t and dC_{h_e}/di_t necessary for application of method of reference 4. $\Delta\alpha_{t_1}$ should be selected as change in angle of attack of the tail in the minimum speed landing attitude, computed by method of reference 4.
			3a	$\delta_e = 0^\circ$		
			3b	$\delta_e = -10^\circ$		
			3c	$\delta_e = -20^\circ$		
			3d	$\delta_e = -25^\circ$		
				$i_t = \text{normal} + \Delta\alpha_{t_1}$		
			4a	$\delta_e = 0^\circ$		
			4b	$\delta_e = -10^\circ$		
			4c	$\delta_e = -20^\circ$		
			4d	$\delta_e = -25^\circ$		Run 5, with tail removed, required for determination of C_{L_α}, used to compute $\Delta\alpha$ and $\Delta\alpha_{t_1}$ by method of reference 4.
			5	Tail off		
III	To determine if elevator balance is sufficient to maintain control forces within required limits.	Critical condition will either be in landing or in accelerated flight with propellers windmilling where stability will be greatest and consequently the stick force per g the highest.		Landing: Data required is same as for II above.	4	
		In landing, a maximum force of 35 pounds for stick-type control and 50 pounds for wheel-type control is permissible (with trim tab set at $1.4V_{stall}$, propeller windmilling).		Accelerated Flight: Polars with propellers windmilling, flaps and gear up.		
		The required stick-force gradient in accelerated flight varies with type of airplane and must be determined from flying-qualities specifications being followed.	6a	$\delta_e = 0^\circ$		Run 7, with increased incidence, made for the purpose of determining dC_m/di_t and dC_{h_e}/di_t necessary for accelerated flight calculations. Value of $\Delta\alpha_{t_2}$ should be determined as maximum induced angle at tail in accelerated flight.
			6b	$\delta_e = -5^\circ$		
			6c	$\delta_e = -10^\circ$		
				$i_t = \text{normal} + \Delta\alpha_{t_2}$		
			7	$\delta_e = 0^\circ$		

TABLE II

Lateral Characteristics
(Single- and Twin-engine Airplanes)

Item	Purpose and requirement	Critical condition	Run No.	Description of Run	Fig. No.	Remarks
IV	To determine if the wing dihedral is great enough to provide at least neutral dihedral effect for the conditions of flight specified.	The critical condition will be in the approach with flaps down and with power on where power and flap effects combine to reduce the dihedral effect. (This condition will normally be worse with ailerons free, but it can be checked to a very good first approximation with ailerons fixed)	 8a 8b 8c	Yaw run at approach attitude with flaps and gear down and 50 percent normal rated power. $\delta_r = 0°, \psi = -30°$ to $30°$ $\delta_r = 20°, \psi = -30°$ to $0°$ $\delta_r = -20°, \psi = 0°$ to $30°$	5	Army calls for stability at $1.2V_{stall}$ (propeller windmilling) with 50 percent rated power. Navy calls for stability in "the approach with considerable power". This condition will normally coincide with the condition outlined for the Army above. The angle of attack for these tests should be chosen on the basis of $C_{L_{max}}$ obtained in the wind tunnel (used in computing $1.2V_{stall}$) but the power (T_c) should be set in accordance with the estimated speed under full scale conditions.
V	To determine if proper balance exists between dihedral effect and directional stability to avoid oscillatory divergence.	Critical condition will be the high speed (clean) condition where dihedral effect will be maximum and directional stability minimum (due to small power effects)	 9	Yaw run at high speed attitude, flaps and gear up, propeller windmilling (or high-speed T_c) $\delta_r = 0°, \psi = -30°$ to $30°$	6	Some doubt exists as to whether or not this criterion expresses a true maximum limit for dihedral. It is believed that an airplane can have dihedral under this limit and yet have an undesirably large roll due to sideslip, and that the tolerable amount actually varies with the type of airplane. However no specific characteristic expressing such a criterion exists.
VI	To determine if ailerons are sufficiently effective to furnish minimum $\left(\frac{pb}{2V}\right)_{max}$ required.	Critical condition will be at low speed (flaps up or down) where aileron effectiveness is usually lowest and reduction in $\left(\frac{pb}{2V}\right)_{max}$ due to yawing is greatest.	 10a 10b 10c 11a 11b 11c	Polar with windmilling propeller. Flaps and gear retracted $\delta_{a_L}=0$, $\delta_{a_R}=0$ $\delta_{a_L}=3/4$ Down, $\delta_{a_R}=3/4$ Up $\delta_{a_L}=$Full Down, $\delta_{a_R}=$Full Up Flaps and gear extended $\delta_{a_L}=0$, $\delta_{a_R}=0$ $\delta_{a_L}=3/4$ Down, $\delta_{a_R}=3/4$ Up $\delta_{a_L}=$Full Down, $\delta_{a_R}=$Full Up	7	For a single-engine airplane runs 11a, b, and c are needed for computations of necessary rudder balance. See Table No. III.
VII	To determine if ailerons are closely enough balanced to furnish required $\left(\frac{pb}{2V}\right)_{max}$ with low enough control forces.	Critical condition will be at highest speed at which requirement applies, normally $.8V_{max}$. Required force and rate of roll varies with type of airplane.		Data required will be furnished by runs 10a, b, and c, supplemented by two-dimensional hinge moment data.	8	For conventional-type ailerons there are normally sufficient two-dimensional data at high Reynolds number which will form a reliable basis for stick-force computations.

TABLE III

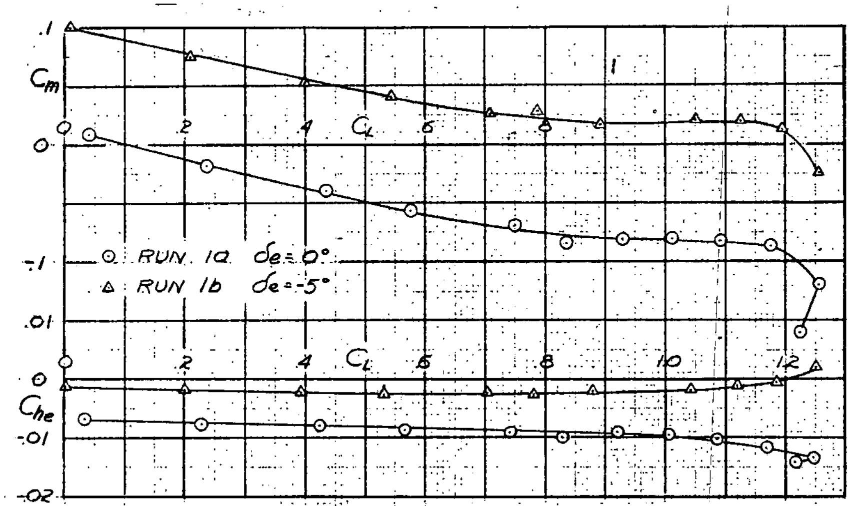
Directional Characteristics
(Single-engine Airplane)

Item	Purpose and requirement	Critical condition	Run No.	Description of run	Fig. No.	Remarks
VIII	To determine if sufficient directional stability is present to avoid rudder force reversal or rudder force reduction at large angles of sideslip.	Critical condition will be at highest angles of sideslip attainable when propeller is operating at a high thrust coefficient. Dependent upon the airplane configuration and power-off stability characteristics, this condition may be critical with flaps either up or down. Both flight conditions should therefore be checked.		Yaw runs at attitude corresponding to $1.2V_{stall}$ (propeller windmilling), flaps and gear up, T_c corresponding to normal rated power.	9	It should be noted that the condition for which the rudder is trimmed will bear an important influence on the rudder reversal characteristics. It is assumed that the incremental tab effects can be estimated and applied to tab-zero data.
			12 a 12 b, c 12 d, e 12 f, g	$\delta_r = 0^\circ$, $\psi = -30^\circ$ to 30° $\delta_r = \pm 15^\circ$ $\delta_r = \pm 20^\circ$ $\delta_r = \pm 25^\circ$ $\psi = 0$ to 30° for $-\delta_r$ $\psi = 0$ to -30° for $+\delta_r$		For airplane being tested for compliance with Army specifications only, the T_c and attitude requirements are less severe and may be changed to the following: Flaps up - T_c of power for level flight Flaps down - attitude of $1.2V_{stall}$ (propeller windmilling); T_c of 50 percent normal rated power
				Yaw runs at attitude corresponding to $1.1V_{stall}$ (propeller windmilling), flaps and gear down, T_c corresponding to take-off power.	10	The above remark also applies to any airplane on which low-speed extreme power handling characteristics are considered of secondary importance.
			13 a 13 b 13 c 13 d	ψ range $= 0^\circ$ to -30° $\delta_r = 0^\circ$ $\delta_r = 15^\circ$ $\delta_r = 20^\circ$ $\delta_r = 25^\circ$		It should be noted that in computation of rudder required to hold steady sideslip, C_n due to aileron has been neglected (figs. 9, 10, 13, 14, and 16).
IX	To determine if the rudder is large enough for necessary control under all normal flight conditions.	Critical condition will be (a) Maintenance of flight with wings level under high thrust conditions of wave-off, where considerable right rudder is required to neutralize effects due to slipstream twist or (b) Maintenance of zero sideslip for above condition with sufficient reserve rudder to compensate for adverse yaw induced by full right aileron deflection.	 14 a 14 b 14 c 14 d	Yaw runs at attitude corresponding to $1.1V_{stall}$ (propeller windmilling), flaps and gear down, T_c corresponding to take-off power. ψ range $= -10^\circ$ to 20° $\delta_r = 0^\circ$ $\delta_r = -15^\circ$ $\delta_r = -20^\circ$ $\delta_r = -25^\circ$	11 and 12	The data required to determine adverse aileron yaw are obtained from Runs 11 a, b, and c. For airplanes being tested for compliance with Army specifications only, analysis may be confined to condition (b), and attitude changed to $1.2V_{stall}$ (propeller windmilling) with T_c for power for level flight. The above remark also applies to any airplane on which low-speed extreme power handling characteristics are considered of secondary importance.
X	To determine if the rudder has sufficient balance to keep the pedal forces within the required 180-pound limit.	Critical condition will occur when attempt is made to perform maneuvers listed under IX above (without aid of a trim-tab adjustment) after an extreme change of power.		Data required will be furnished by runs of item IX above. Original rudder trim with minimum power assumed to exist with tab neutral.	11 and 12	The condition specified will normally give higher pedal forces than will be encountered in a terminal velocity dive provided the rudder is assumed to be trimmed for high-speed flight before the dive is entered. Assumption of less favorable original rudder trim may make the dive condition critical.

TABLE IV

Directional Characteristics
(Twin-engine Airplanes)

Item	Purpose and requirement	Critical condition	Run No.	Description of run	Fig. No.	Remarks
VIII	To determine if sufficient directional stability is present (a) to avoid rudder pedal force reversals or reduction at large angles of sideslip. and	(a) Critical condition will be at highest anglesof right sideslip attainable when the propeller is operating at a high-thrust coefficient.	 12a 12b 12c 12d	(a) Yaw run at approach attitude, with flaps and gear down, T_c as called for by requirement (both engines operating) Yaw range = 0 to -30° $\delta_r = 0^{\circ}$ $\delta_r = 10^{\circ}$ $\delta_r = 20^{\circ}$ $\delta_r = 25^{\circ}$	13	(a) Army calls for rudder-free directional stability at $1.2V_{stall}$ (propeller windmilling) with 50 percent rated power and tab set for trim at zero sideslip. C_{h_r} for tab can be estimated. Navy calls for no reduction of rudder pedal force as the angle of sideslip is increased, with take-off power and neutral trim tab. As the Navy does not give a definite minimum speed, $1.1V_{stall}$ (propeller windmilling) is assumed to be the lowest speed at which this requirement need be met. Runs in right sideslip are called for since normally this will represent a more extreme condition than left sideslip.
	(b) To permit the airplane to be balanced directionally in steady flight, with rudder free and asymmetric power by banking to a moderate angle.	(b) The critical condition will be represented by the failure of one engine shortly after take-off.	 13a 13b	(b) Yaw run at attitude corresponding to $1.2V_{stall}$, flaps at take-off setting, gear down. Take-off power on one engine; other engine, propeller windmilling. Make runs with the rudder free and the ailerons set with full deflection in direction to bring wing with dead engine up. Right engine operating $\psi = 0$ to -30° Left engine operating $\psi = 0$ to 30°	14	(b) This requirement is not called for by the Army. Navy specifications require angle of bank to be limited to 15°, 25°, or 35° depending on type of airplane.
IX	To determine if the rudder is capable of maintaining the required control under all conditions of steady flight.	Critical condition will be after single-engine failure where the rudder control should be powerful enough to (a) hold zero sideslip at all speeds down to 120 percent of the stalling speed in the clean condition. or	 14a 14b 14c 14d	(a) Yaw runs at attitude corresponding to $1.2V_{stall}$, flaps and gear up. Take-off power on right engine; left engine, propeller windmilling. Yaw range = -20° to 10° $\delta_r = 0^{\circ}$ $\delta_r = -10^{\circ}$ $\delta_r = -20^{\circ}$ $\delta_r = -25^{\circ}$	15	(a) This requirement applies only to Navy airplanes.
		(b) Hold at least 10° of sideslip at 120 percent of the stalling speed in the take-off condition.	 15a 15b 15c 15d	(b) Yaw runs at take-off attitude ($1.2V_{stall}$) flaps in take-off position, gear down. Take-off power on right engine; left engine, propeller windmilling. Yaw range + 5 to -25° $\delta_r = 0^{\circ}$ $\delta_r = -10^{\circ}$ $\delta_r = -20^{\circ}$ $\delta_r = -25^{\circ}$	16	(b) This requirement applies only for Army airplanes.
X	To determine if the rudder has enough balance to keep the rudder pedal forces within the 180-pound limit.	Critical condition will be in the flight condition (a) and (b) listed above.		(a) Data required is obtained from Run 13 above. (b) Data required is obtained from Run 14 above.	15 and 16	(a) Most severe requirement applied by the Navy. (with respect to rudder pedal forces) (b) Requirement (b) is usually less severe than (a) but is the most severe applied by the Army.



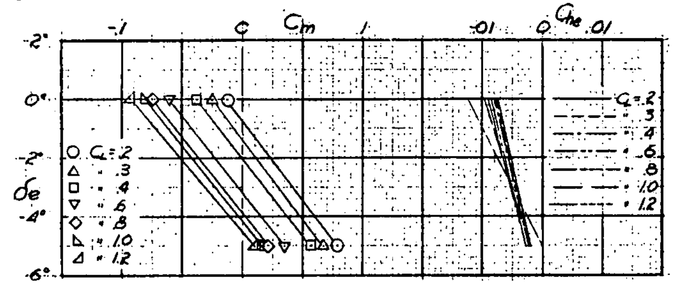
(A) MODEL CHARACTERISTICS DETERMINED FROM WINDTUNNEL TESTS
C.G. AT 26-PERCENT MAC

①	②	③	④	⑤	⑥	⑦	⑧
		NEUTRAL TAB					
C_L	V_i IN mph	δ_e FOR C_m EQUALS ZERO FROM CROSSPLOT	C_{he} FOR ③ FROM CROSSPLOT	C_{he} FOR ③ AT BEST CLIMB SPEED (C_L = .4)	TAB C_{he}[1] WITH TAB SET FOR TRIM AT BEST CLIMB SPEED	C_{he} WITH TRIM AT BEST CLIMB SPEED	STICK FORCE FROM $F/C_{he}q = 23.8$
.2	252	-.6° (UP)	-.0065	-.0057	+.0057 (1)	-.0008	3.1 (PUSH)
.3	206	-1.3° "	-.0063	——	" "	-.0006	1.6 "
.4	178	-1.9° "	-.0057	——	" "	0	0 ——
.6	146	-3.1° "	-.0047	——	" (1.04)	+.0012	1.6 (PULL)
.8	126	-4.0° "	-.0032	——	" (1.10)	+.0031	3.0 "
1.0	113	-4.2° "	-.0027	——	" (1.19)	+.0041	3.2 "
1.2	103	-4.4° "	-.0006	——	" (1.32)	+.0069	4.9 "

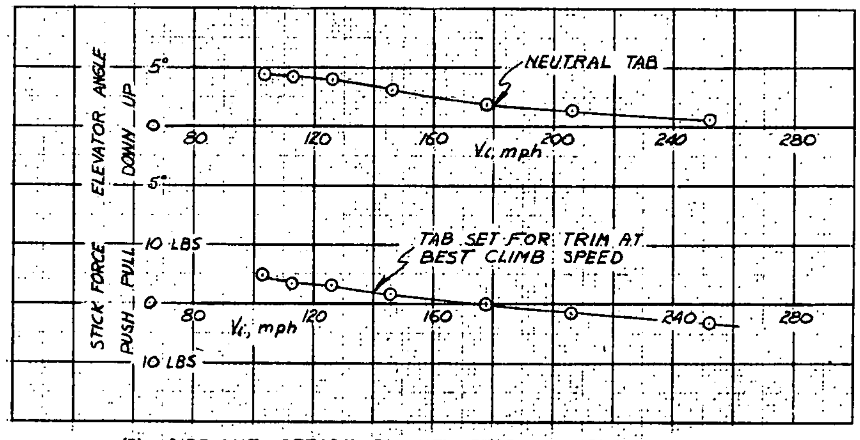
1 THE VARIATION IN C_{he} IN COLUMN ⑥ IS DUE TO THE VARIATION OF q_t/q AT THE TAIL. THE FACTOR INVOLVED, GIVEN IN COLUMN ⑥, MAY BE OBTAINED FROM THE RATIOS OF $dC_m/d\delta_e$ IN FIGURES 1(C) AND 4(C) WITH DUE REGARD FOR THE RELATIVE LOCATION OF THE TAB AND SLIPSTREAM

WING LOADING = 32.6 LB/SQ FT

(B) COMPUTATION TABLE

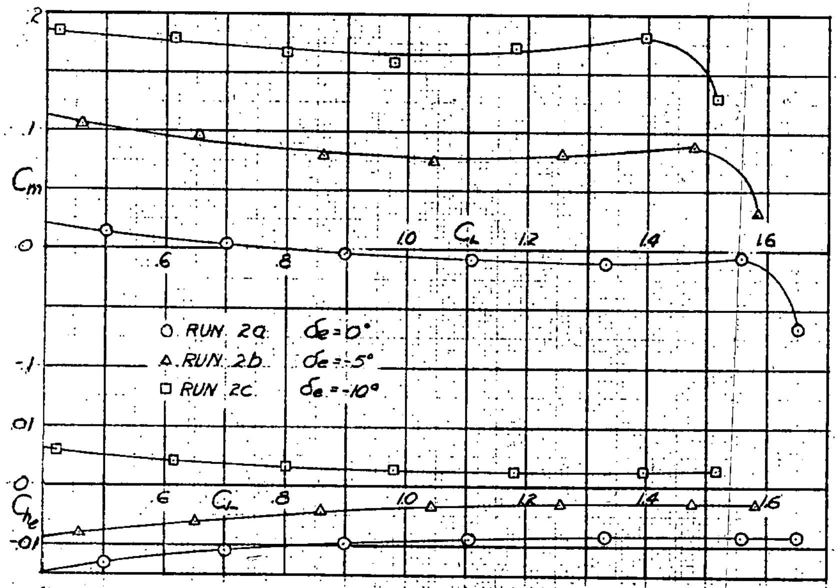


(C) MODEL CHARACTERISTICS CROSSPLOTTED



(D) AIRPLANE STEADY FLIGHT CHARACTERISTICS

FIGURE 1.- VARIATION OF ELEVATOR ANGLE AND STICK FORCE WITH SPEED. STEADY FLIGHT WITH FLAPS AND GEAR UP AND RATED POWER. SINGLE-ENGINE AIRPLANE.



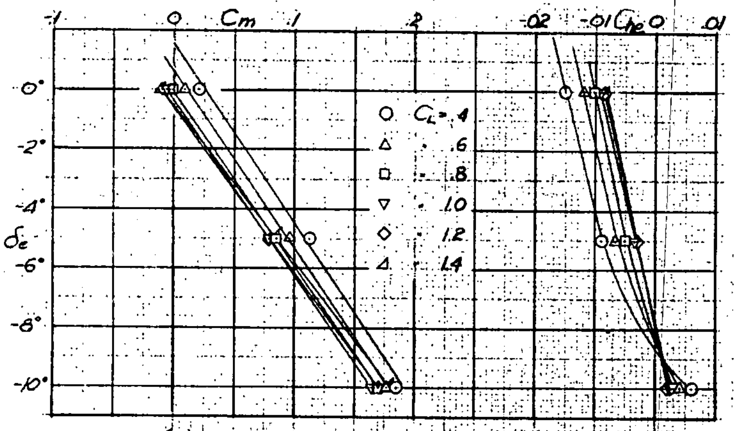
(A) MODEL CHARACTERISTICS DETERMINED FROM WINDTUNNEL TESTS. C.G. AT 26-PERCENT MAC

①	②	③	④	⑤	⑥	⑦	⑧
		NEUTRAL TAB					
C_L	V_i I.N. mph	δ_e FOR C_m EQUALS ZERO FROM CROSSPLOT	C_{he} FOR ③ FROM CROSSPLOT	C_{he} FOR ③ AT 1.4 V_{STALL} (C_L = .92) FROM CROSSPLOT	TAB C_{he} WITH TAB SET FOR TRIM AT 1.4 V_{STALL}	C_{he} WITH TRIM AT 1.4 V_{STALL}	STICK FORCE FROM $F/C_{he}\,q$ = 23.8
.4	178	1.5°	-.0170	-.0095	.0095[1]	-.0075	14.5 (PUSH)
.6	146	.6°	-.0130	—	"	-.0035	4.5 "
.8	126	0°	-.0102	—	"	-.0007	.7 "
1.0	113	-.4°	-.0088	—	"	+.0007	.5 (PULL)
1.2	103	-.6°	-.0080	—	"	+.0015	1.0 "
1.4	94	-.6°	-.0076	—	"	+.0019	1.1 "

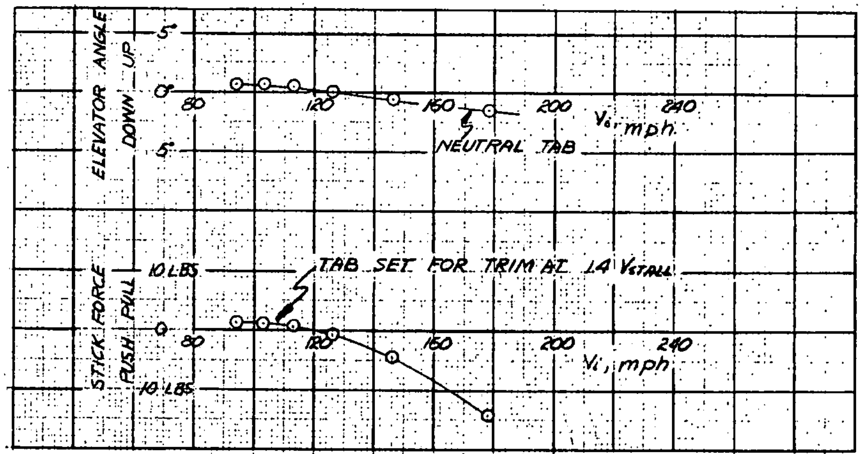
[1] NO ALLOWANCE WAS MADE FOR q_t/q AT THE TAIL WITH RESPECT TO TAB C_{he}. IF NECESSARY IT CAN BE DONE BY THE THE METHOD IN FIGURE 1 (B).

WING LOADING = 32.6 LB/SQ. FT.

(B) COMPUTATION TABLE.

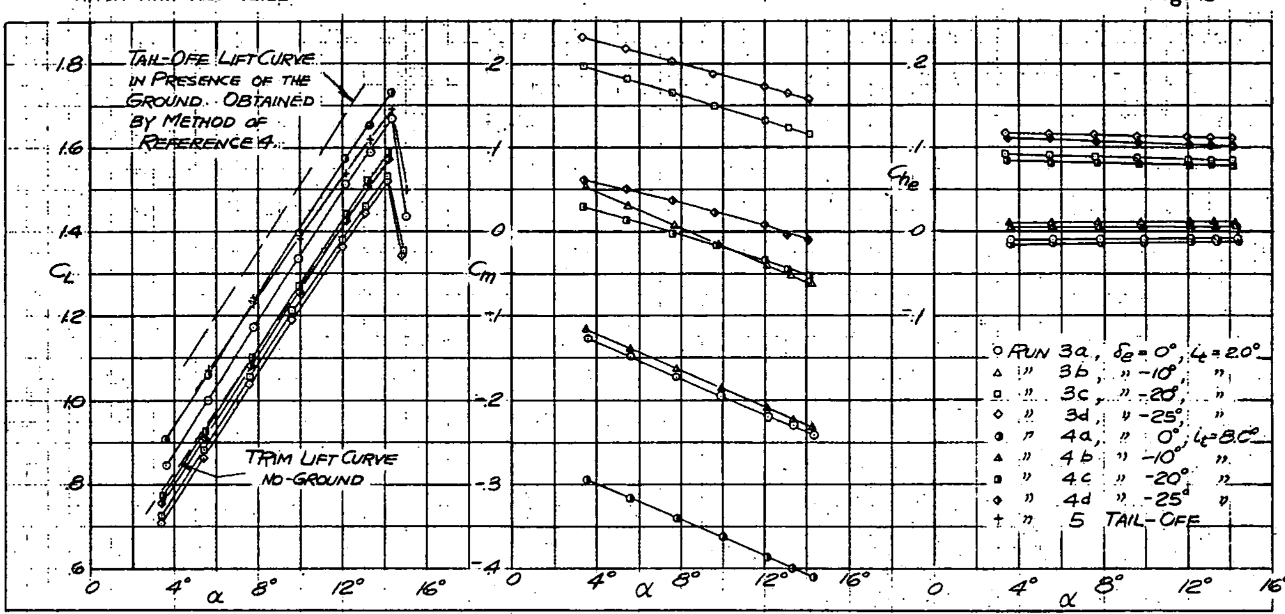


(C) MODEL CHARACTERISTICS CROSSPLOTTED



(D) AIRPLANE STEADY FLIGHT CHARACTERISTICS

FIGURE 2.-VARIATION OF ELEVATOR ANGLE AND STICK FORCE WITH SPEED. STEADY FLIGHT WITH FLAPS AND GEAR DOWN, 30 PERCENT NORMAL RATED POWER. SINGLE-ENGINE AIRPLANE.



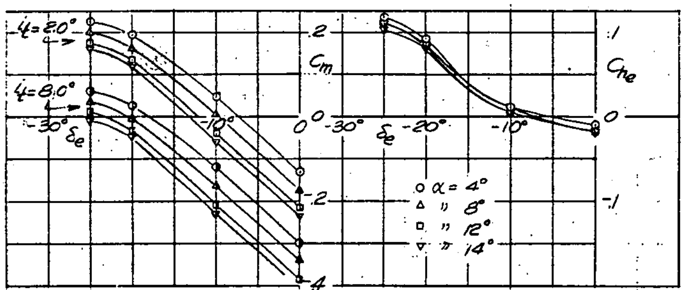
(A) MODEL CHARACTERISTICS DETERMINED FROM WIND TUNNEL TESTS – c.g. AT 16-PERCENT MAC

①	②	③	④	⑤	⑥	⑦	⑧	⑨	⑩	⑪	⑫	⑬	⑭	⑮	⑯	⑰
TAB NEUTRAL												TAB C_{h_e} TO TRIM AT 1.4 V_{STALL}, NO GROUND. NORMAL i_t				
α, ANGLE OF ATTACK OF REFERENCE LINE.	C_{L_W} IN PRESENCE OF GROUND CORRESPONDING TO α OF ① FROM PART (A)	ϵ, DOWNWASH ANGLE IN PRESENCE OF GROUND CORRESPONDING TO C_{L_W} OF ② AND α OF ①. OBTAINED FROM REF. 4.	TAIL OFF α WITH NO GROUND EFFECT CORRESPONDING TO C_{L_W} OF ② FROM PART (A).	ϵ, DOWNWASH ANGLE WITH NO GROUND EFFECT CORRESPONDING TO C_{L_W} OF ② AND α OF ④. OBTAINED FROM REF. 4.	$\Delta\epsilon_t$, INCREMENT OF DOWNWASH ANGLE AT TAIL DUE TO GROUND EFFECT, EQUALS ϵ OF ③ MINUS ϵ OF ⑤.	$\Delta\alpha$ DUE TO GROUND EFFECT EQUALS ① MINUS ④ AS OBTAINED FROM REF. 4	$\Delta\alpha_{t_t}$, THE TOTAL CHANGE IN TAIL ANGLE OF ATTACK RESULTING FROM GROUND EFFECT EQUALS $\Delta\alpha$ OF ⑦ MINUS $\Delta\epsilon_t$ OF ⑥.	δ_e FOR C_m = 0, WITH TAIL INCIDENCE INCREASED BY INCREMENT EQUAL TO $\Delta\alpha_{t_t}$ OF ⑧ AND WITH ANGLE OF ATTACK EQUAL TO α OF ④. OBTAINED FROM PART (C).	C_L CORRESPONDING TO α OF ④ FROM TRIM LIFT CURVE OF PART (A).	V_i FOR C_L OF ⑩. WING LOADING = 25 LB/SQ.FT.	C_{h_e} CORRESPONDING TO α OF ④, $\Delta\alpha_{t_t}$ OF ⑧, AND δ_e OF ⑨ FROM PART (C).	δ_e FOR C_m = 0, NO GROUND, AT 1.4 V_{STALL} FROM PART (C). (C_L = .9, α_{TRIM} = 4.9° FROM PART (A))	C_{h_e} FOR δ_e AND α_{TRIM} OF ⑬, FROM PART (C).	TAB C_{h_e} EQUALS – C_{h_e} OF ⑭	C_{h_e} IN LANDING EQUALS ⑫ + ⑮	STICK FORCE FROM F/qC_{h_e} = 23.8
11°	1.59	3.3°	12.9°	11.2°	–7.9°	–1.9°	6.0°	–25°	1.48	81.4 mph	.105	–8.0°	.005	–.005	.100	41.4 LB. PULL

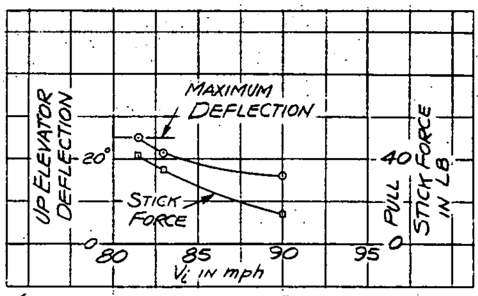
THE ABOVE COMPUTATIONS ARE FOR THE THREE POINT ATTITUDE. COMPUTATIONS FOR LANDING AT GREATER SPEEDS ARE MADE BY INTERPOLATING BETWEEN THE i_t LIMITS OF PART (A).

THE RESULTS OF REFERENCE 5 INDICATE A SMALLER INCREASE IN C_{L_W} AT A CONSTANT ATTITUDE, DUE TO GROUND EFFECT, THAN THAT COMPUTED BY REFERENCE 4. REFERENCE 5 ALSO INDICATES A PITCHING MOMENT INCREMENT ON THE WING, DUE TO GROUND EFFECT THAT TENDS TO STALL THE AIRPLANE. THE COMPUTATIONS ABOVE DO NOT ALLOW FOR THE GROUND EFFECTS NOTED IN REFERENCE 5. THIS PROCEDURE RESULTS IN A CONSERVATIVE ESTIMATE OF δ_e AND STICK FORCE TO LAND (WITH RESPECT TO REF. 5).

(B) COMPUTATION TABLE.

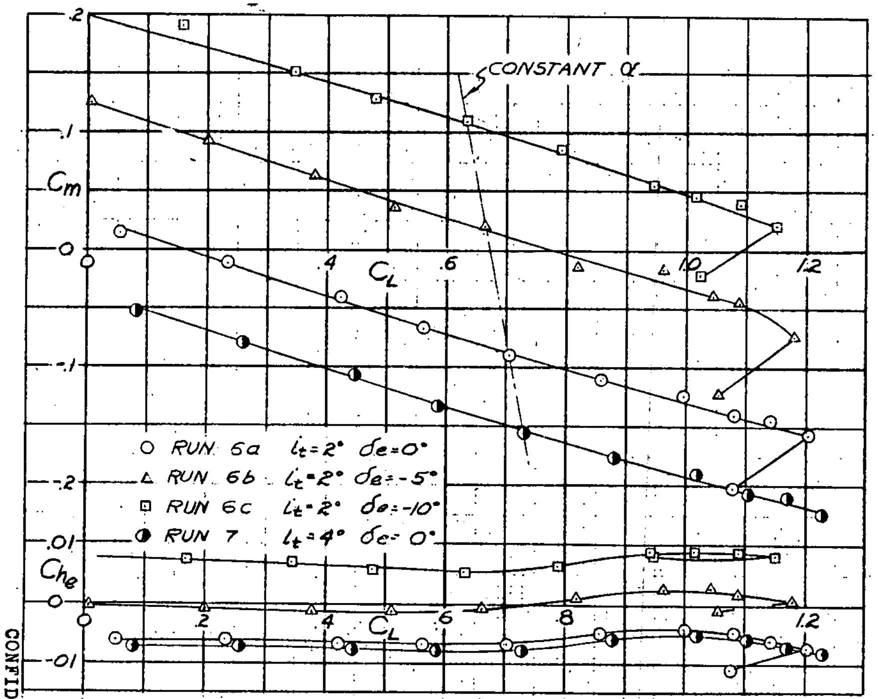


(C) MODEL CHARACTERISTICS CROSSPLOTTED



(D) AIRPLANE LANDING CHARACTERISTICS

FIGURE 3.- VARIATION OF ELEVATOR ANGLE AND STICK FORCE IN LANDING. FLAPS AND GEAR DOWN, PROPELLOR WINDMILLING. SINGLE-ENGINE AIRPLANE.



(A) MODEL CHARACTERISTICS DETERMINED FROM WINDTUNNEL TESTS. C.G. AT 26-PERCENT MAC.

①	②	③	④	⑤	⑥	⑦	⑧	⑨	⑩	⑪	⑫	⑬
						NEUTRAL TAB						
n	C_L FOR VALUES OF ①	δ_e TO BALANCE C_m DUE TO STATIC STABILITY FROM CROSSPLOT	$\frac{n^2-1}{n}$	$\Delta\alpha_t$ DUE TO CURVILINEAR FLIGHT	$\Delta\delta_e$ TO BALANCE C_m DUE TO $\Delta\alpha_t$	δ_e FOR $C_m=0$ IN CURVILINEAR FLIGHT EQUALS ③ PLUS ⑥	C_{he} FOR ② AND ⑦ FROM CROSSPLOT	C_{he} DUE TO $\Delta\alpha_t$	C_{he} IN CURVILINEAR FLIGHT EQUALS ⑧ PLUS ⑨	TAB C_{he} WITH TAB SET FOR TRIM AT n OF 1	C_{he} IN CURVILINEAR FLIGHT WITH TRIM AT n OF 1 EQUALS ⑩+⑪	STICK FORCE FROM $F/C_{he}\,q = 23.8$
1	.148	.1°	0	0°	0°	.1°	-.0062	0	-.0062	+.0062	0	0
2	.296	-1.1°	3/2	.31°	-.54°	-1.6°	-.0048	-.0002	-.0050	"	.0012	6.3
3	.444	-2.2°	8/3	.55°	-.96°	-3.2°	-.0034	-.0003	-.0037	"	.0025	13.1
4	.592	-3.5°	15/4	.77°	-1.34°	-4.9°	-.0018	-.0004	-.0022	"	.0040	21.0
5	.740	-4.8°	24/5	1.03°	-1.79°	-6.6°	+.0018	-.0005	+.0013	"	.0075	39.4
6	.888	-5.9°	35/6	1.20°	-2.09°	-8.0°	+.0055	-.0006	+.0049	"	.0111	58.0

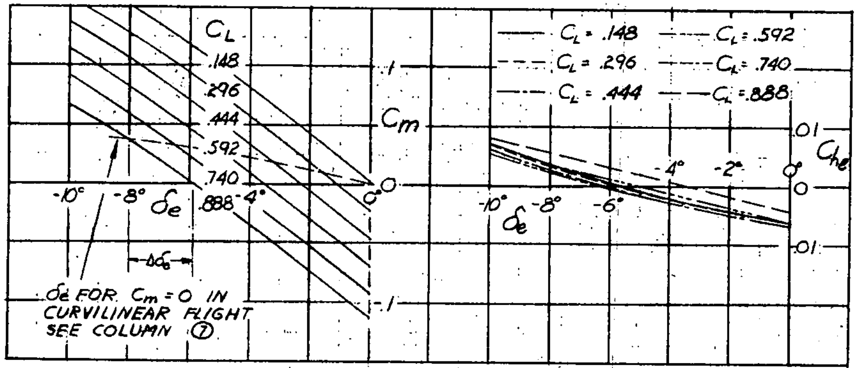
$$\Delta\alpha_t = \frac{l_H\, g}{(V_{fps})^2}\left(\frac{n^2-1}{n}\right)57.3 = .206\left(\frac{n^2-1}{n}\right)$$

$$⑨ = \Delta\alpha_t\left(\frac{dC_{he}}{d i_t}\right)_\alpha = -.0005\,\Delta\alpha_t$$

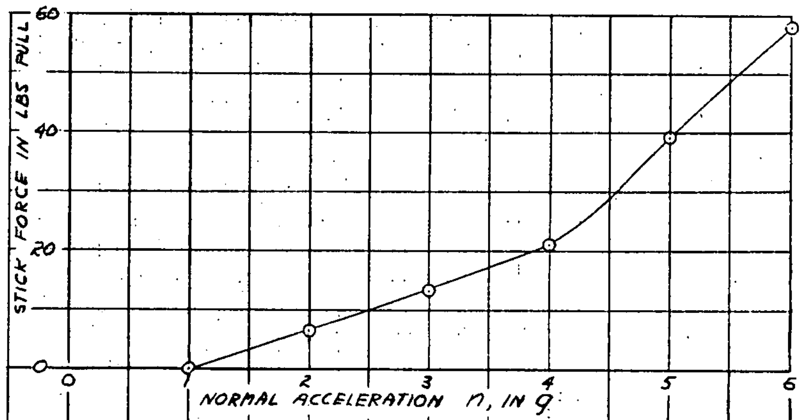
$$\Delta\delta_e = \Delta\alpha_t\Big/\left(\frac{\partial\alpha_t}{\partial\delta_e}\right)_{C_{L_t}} = -\Delta\alpha_t\Big/\frac{\left(\frac{dC_m}{d\delta_e}\right)_\alpha}{\left(\frac{dC_m}{d i_t}\right)_\alpha} = -\frac{\Delta\alpha_t}{.575}$$

$V_i = 294$ mph AT 9,000 FT., WING LOADING = 32.6 LB/SQ.FT.

(B) COMPUTATION TABLE



(C) MODEL CHARACTERISTICS CROSSPLOTTED



(D) AIRPLANE ACCELERATED FLIGHT STICK FORCE CHARACTERISTICS

FIGURE 4.- VARIATION OF ELEVATOR STICK FORCE WITH NORMAL ACCELERATION IN STEADY TURNING FLIGHT. FLAPS AND GEAR UP, PROPELLOR WINDMILLING. SINGLE-ENGINE AIRPLANE.

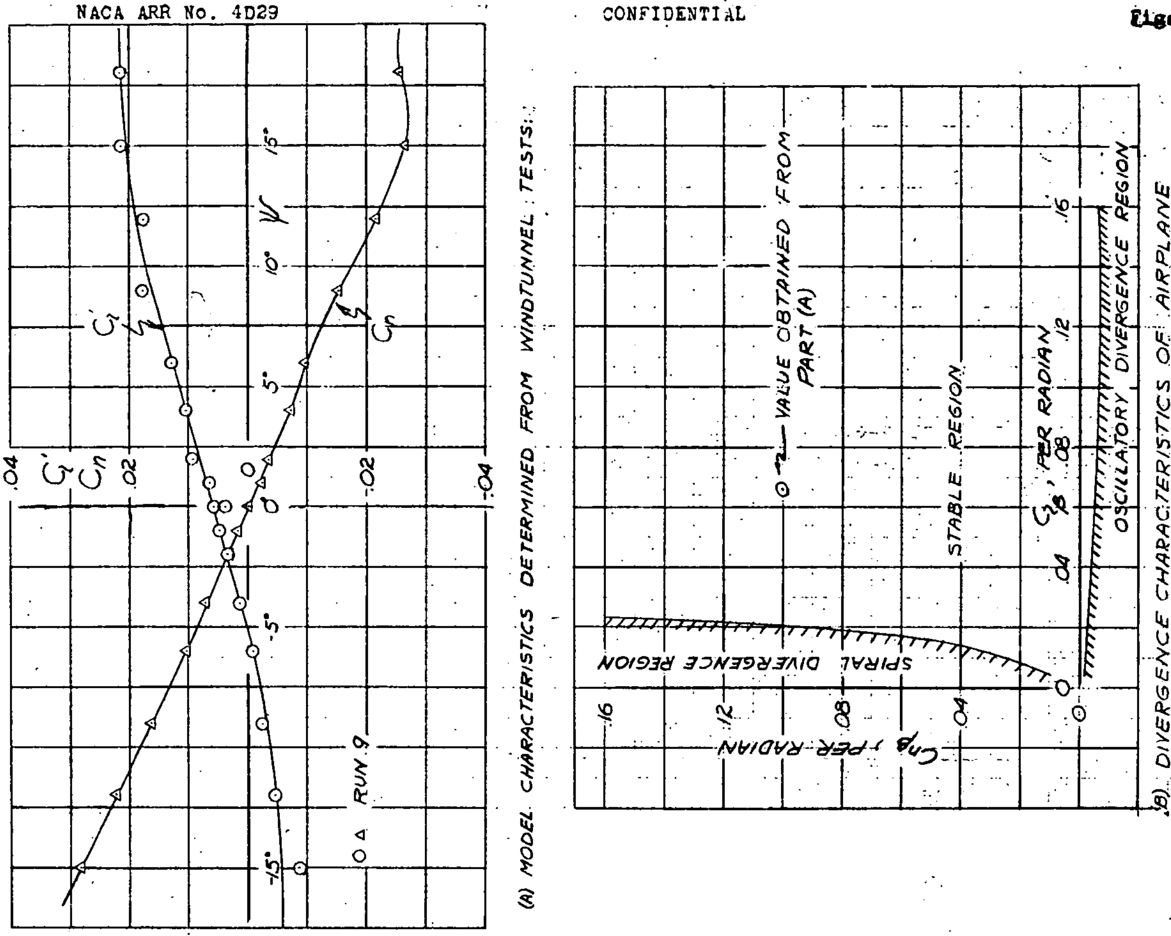


FIGURE 6.- LATERAL-STABILITY CHARACTERISTICS AT HIGH SPEED, FLAPS AND GEAR UP, RATED POWER. SINGLE-ENGINE AIRPLANE.

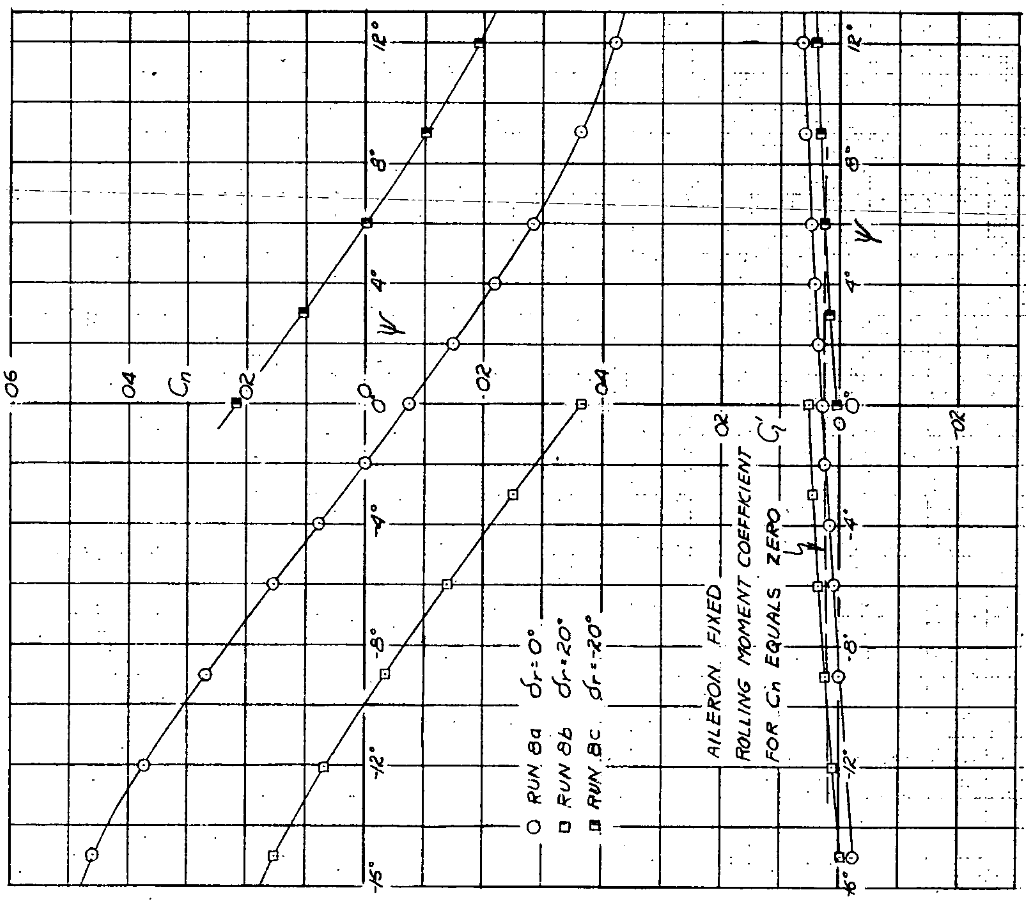


FIGURE 5.- DIHEDRAL CHARACTERISTICS AT LOW SPEED. FLAPS AND GEAR DOWN, 50 PERCENT NORMAL RATED POWER. SINGLE-ENGINE AIRPLANE.

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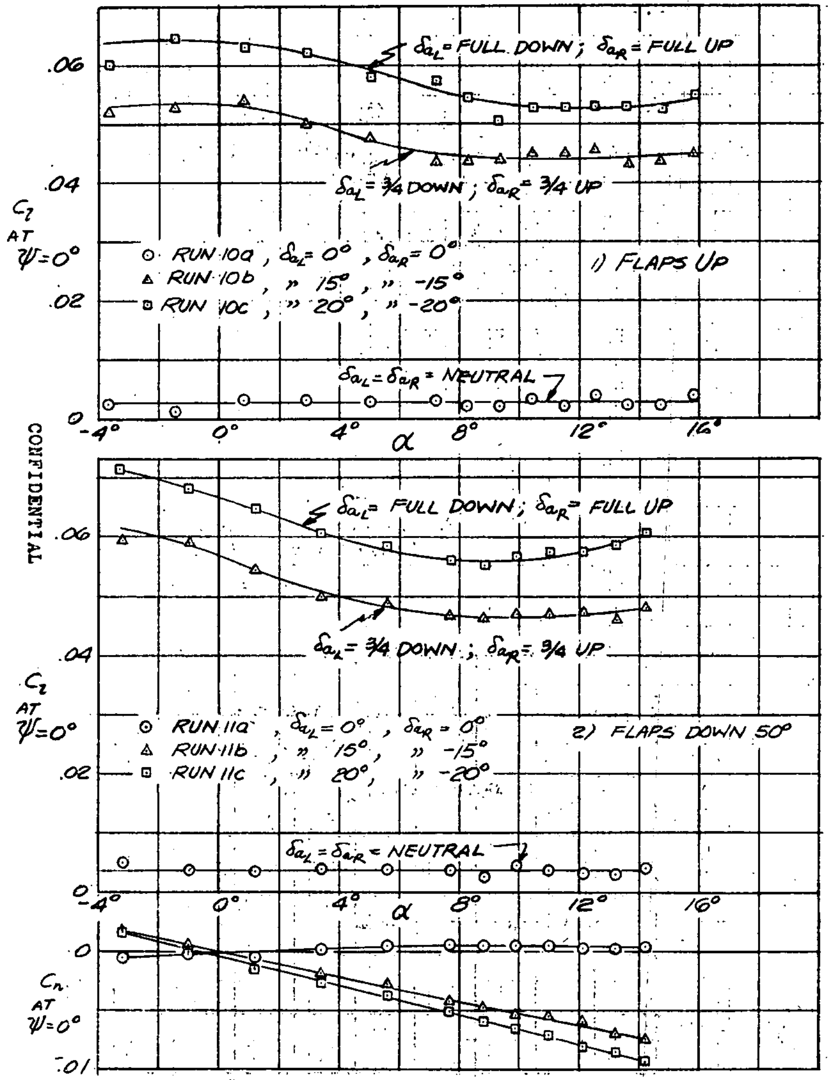
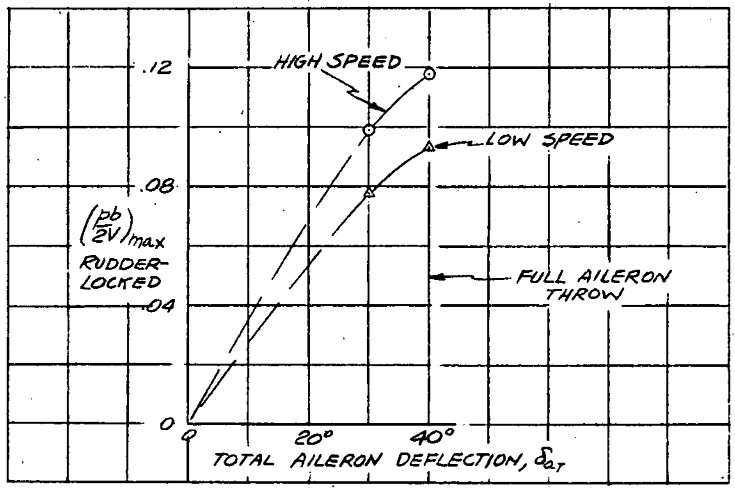


(A) MODEL CHARACTERISTICS DETERMINED FROM WIND TUNNEL TESTS

① CONDITION	② AILERON THROW	③ ROLLING MOMENT DUE TO AILERONS FROM FIG (A-1)	④ $\left(\frac{pb}{2V}\right)_{max}$, ZERO SIDESLIP, EQUALS $-\frac{C_{l_a}}{C_{l_p}} = -\frac{③}{C_{l_p}}$ [1]	⑤ $\left(\frac{pb}{2V}\right)_{max}$ REDUCTION FACTOR — DUE TO SIDESLIP [2]	⑥ $\left(\frac{pb}{2V}\right)_{max}$ RUDDER-LOCKED. ④ × ⑤	⑦ TOTAL AILERON DEFLECTION, δ_{a_T}
HIGH-SPEED $V_i = 0.8\,V_{max}$ = 266 mph $C_L = 0.18$ $\alpha = 0.2°$	3/4	.051	.109	.91	.099	30°
	FULL	.061	.130	.91	.118	40°
LOW-SPEED $1.2\,V_{stall}$ $C_L = 0.86$ $\alpha = 10.4°$	3/4	.042	.097	.80	.078	30°
	FULL	.050	.116	.80	.093	40°

1. $C_{l_p} = -0.47$ AT HIGH SPEED AND -0.43 AT LOW SPEED (FROM REF 6)
2. BELIEVED TO BE REPRESENTATIVE OF MODERN HIGH PERFORMANCE AIRPLANES. (ASSUMES A RIGID WING)

(B) COMPUTATION TABLE, FLAPS-UP



(C) AIRPLANE ROLLING CHARACTERISTICS WITH FLAPS UP

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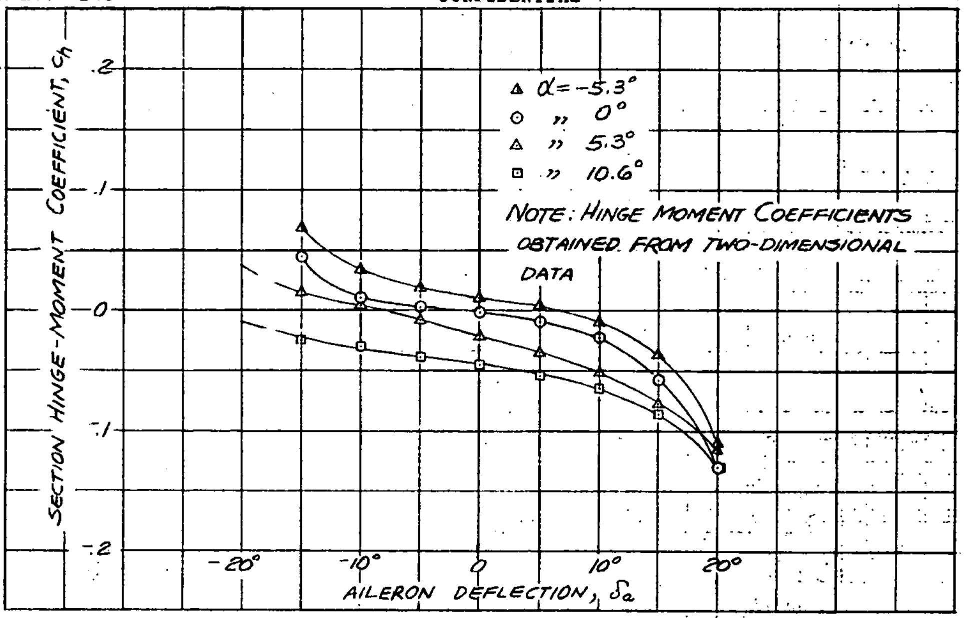
FIGURE 7.- AILERON CONTROL CHARACTERISTICS (pb/2V VS AILERON DEFLECTION). FLAPS AND GEAR UP. SINGLE-ENGINE AIRPLANE.



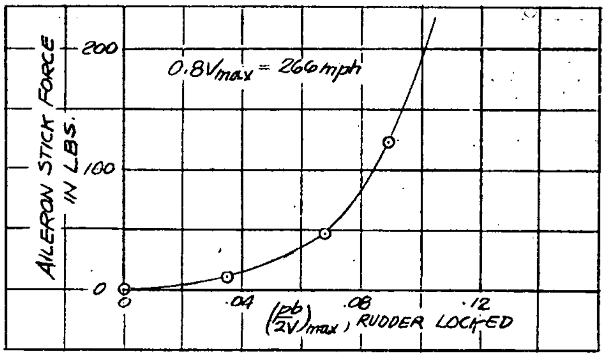
(A) HINGE MOMENT CHARACTERISTICS DETERMINED FROM WIND TUNNEL TESTS

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)
AILERON POSITION	LEFT AILERON DEFLECTION δ_{aL}	RIGHT AILERON DEFLECTION δ_{aR}	$(\frac{pb}{2V})_{max}$, FROM FIG. 7	INDUCED ANGLE DUE TO ROLLING $\Delta\alpha = 40' \times$ (4) [1]	AVERAGE ANGLE OF ATTACK OVER EACH AILERON[2]: LEFT $\alpha_L = \alpha - \Delta\alpha$	RIGHT $\alpha_R = \alpha + \Delta\alpha$	C_{haL} FOR δ_{aL} AND α_L FROM (A) ABOVE	C_{haR} FOR δ_{aR} AND α_R FROM (A) ABOVE	SUMMATION C_{ha} (8) − (9)	STICK FORCE / $f\,C_{ha}$	AILERON CONTROL FORCE IN LBS $f =$ (11) × (10)
0	0	0	0	0	0	0	-.002	-.002	0	12.3	0
1/4 THROW	5°	-5°	.035	1.4°	-1.4°	1.4°	-.005	0	-.005		11
1/2 THROW	10°	-10°	.068	2.7°	-2.7°	2.7°	-.015	.006	-.021		47
3/4 THROW	15°	-15°	.089	3.6°	-3.6°	3.6°	-.044	.021	-.065		123
FULL THROW	20°	-20°	.118	4.7°	-4.7°	4.7°	-.118	.041	-.159		354

[1] $\Delta\alpha = 40' \times (\frac{pb}{2V})_{max} = \frac{l_1 + l_2}{b} \times (\frac{pb}{2V})_{max} \times 57.3$ WHERE l_1 AND l_2 ARE DISTANCES FROM PLANE OF SYMMETRY TO INBOARD AND OUTBOARD ENDS OF THE AILERON

[2] $\alpha = 0.2°$ AT $0.8 V_{max} = 266$ mph; $q = 181$ LB/SQ FT

(B) COMPUTATION TABLE



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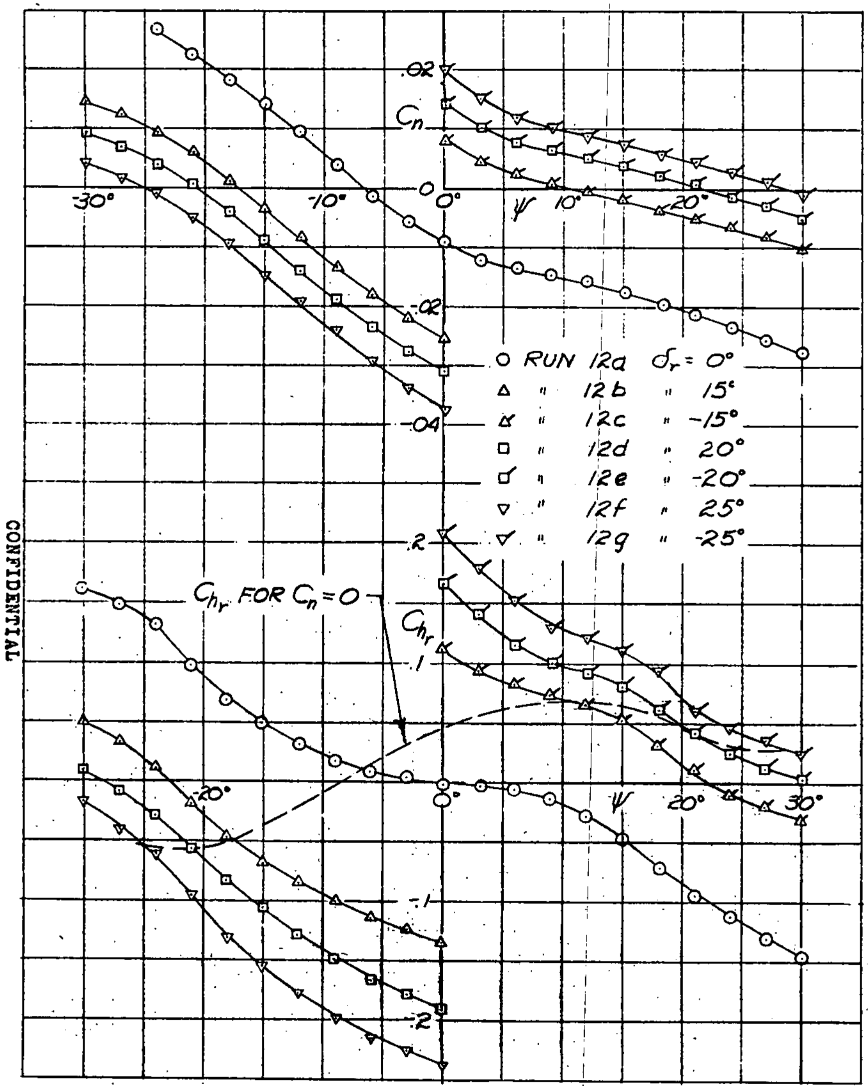
(C) AILERON STICK FORCE CHARACTERISTICS IN STEADY ROLLS

FIGURE 8.- VARIATION OF AILERON STICK FORCE WITH pb/2V AT HIGH SPEED. FLAPS AND GEAR UP. SINGLE-ENGINE AIRPLANE.

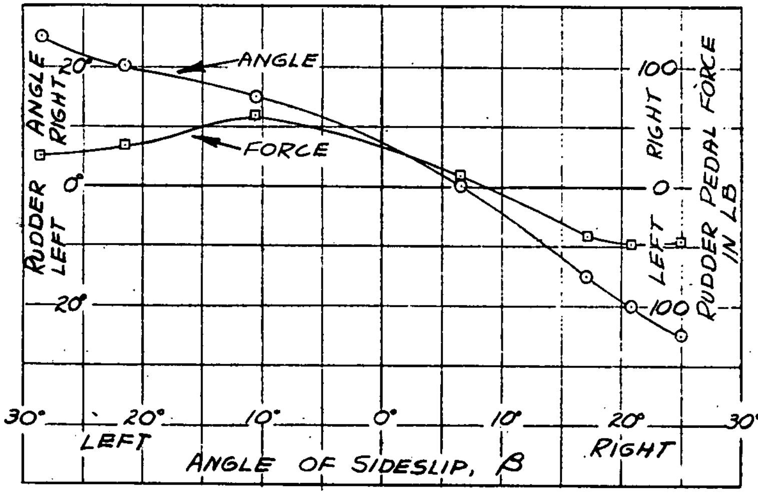


(A) MODEL CHARACTERISTICS DETERMINED FROM WINDTUNNEL TESTS.

① RUDDER ANGLE δ_r	② ψ FOR C_n EQUALS ZERO	③ β FOR $C_n = 0$ EQUALS $-\psi$ FOR $C_n = 0$	④ C_{h_r} FOR ① AND ②	⑤ PEDAL FORCE / $q \cdot C_{h_r}$	⑥ PEDAL FORCE IN LB
-25° R	28.5°	-28.5° L	.030	26.6	25 R
-20° R	21.5°	-21.5° L	.040	"	34 R
-15° R	10.5°	-10.5° L	.070	"	59 R
0°	-6.7°	6.7° R	.010	"	8 R
15° L	-17.2°	17.2° R	-.050	"	42 L
20° L	-20.7°	20.7° R	-.057	"	48 L
25° L	-25.0°	25.0° R	-.055	"	46 L

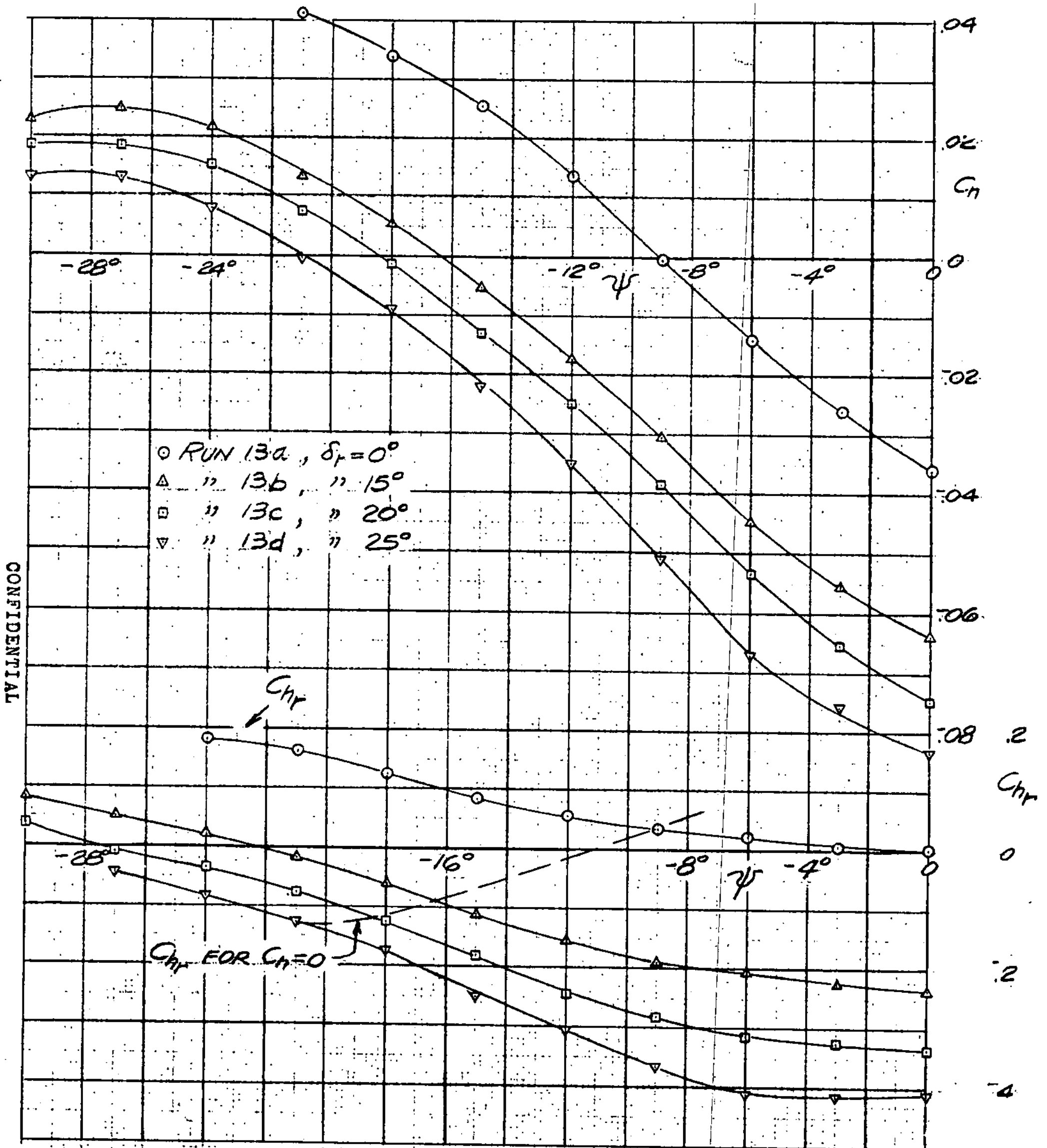
V_i = 111 mph, q = 31.5 LB./SQ.FT., WING LOADING = 32.5 LB/SQ.FT.

(B) COMPUTATION TABLE.



(C) AIRPLANE STEADY SIDESLIP CHARACTERISTICS.

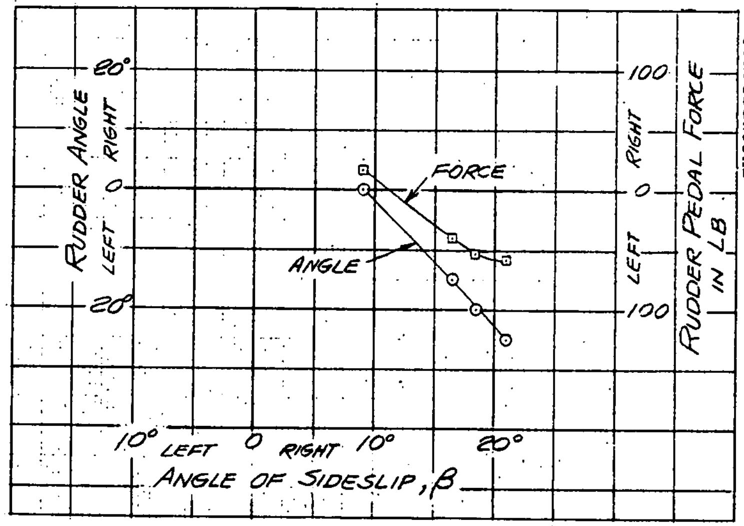
FIGURE 9.-VARIATION OF RUDDER ANGLE AND PEDAL FORCE WITH SIDESLIP AT LOW SPEED. FLAPS AND GEAR UP, NORMAL RATED POWER. SINGLE-ENGINE AIRPLANE.



(A) MODEL CHARACTERISTICS DETERMINED FROM WIND-TUNNEL TESTS

①	②	③	④	⑤	⑥
RUDDER ANGLE δ_r	ψ FOR C_n EQUALS ZERO	β FOR C_n=0 EQUALS $-\psi$ FOR C_n=0	C_{h_r} FOR ① AND ②	PEDAL FORCE / $q \times C_{h_r}$	PEDAL FORCE IN LB
0	-9.1°	9.1°R	.036	26.6	16 R
15°L	-16.4°	16.4°R	-.090	"	40L
20°L	-18.4°	18.4°R	-.119	"	53L
25°L	-20.9°	20.9°R	-.130	"	58L
V_i = 81 mph, q = 16.8 LB/SQ FT, WING LOADING = 32.6 LB/SQ FT					

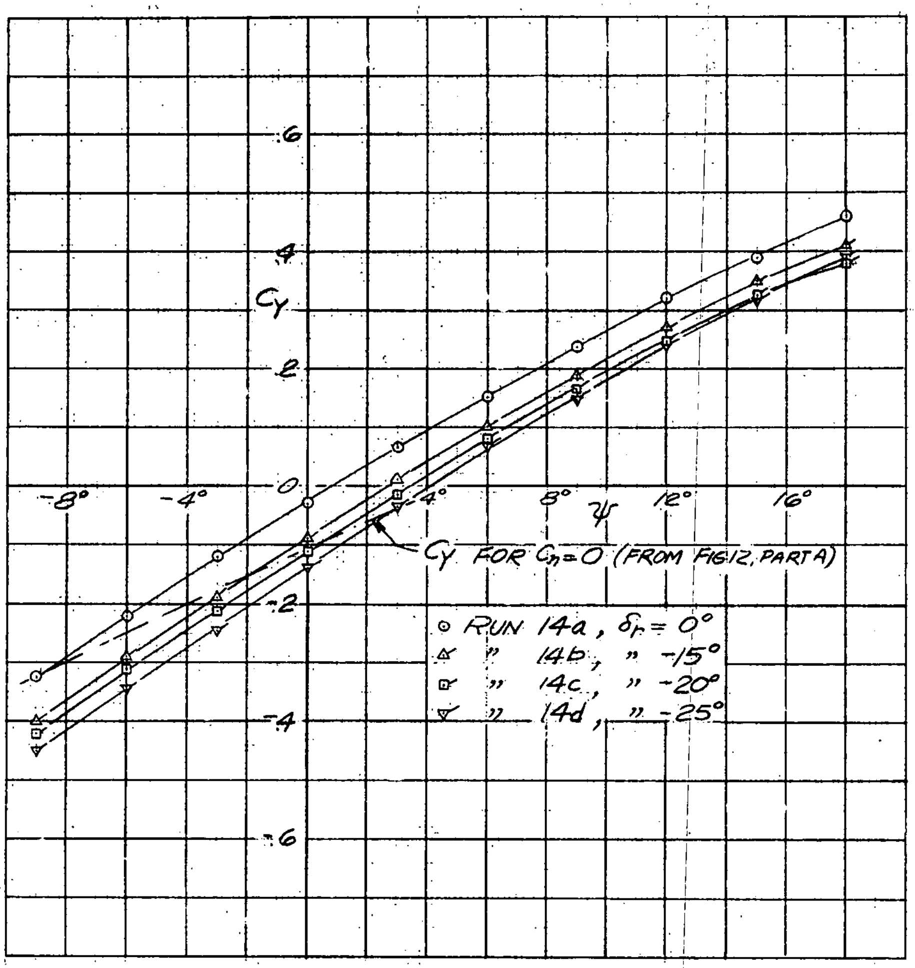
(B) COMPUTATION TABLE



(C) AIRPLANE STEADY SIDESLIP CHARACTERISTICS

RESTRICTED

FIGURE 10.- VARIATION OF RUDDER ANGLE AND PEDAL FORCE WITH SIDE-SLIP IN WAVE-OFF. FLAPS AND GEAR DOWN, TAKE-OFF POWER. SINGLE-ENGINE AIRPLANE.

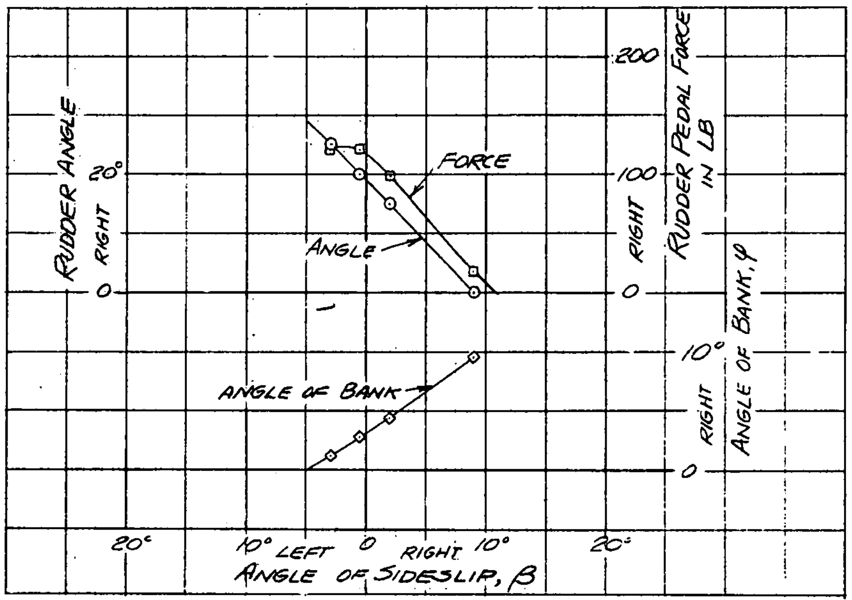


(A) MODEL CHARACTERISTICS DETERMINED FROM WIND-TUNNEL TESTS

① RUDDER ANGLE δ_r	② ψ FOR C_n EQUALS ZERO	③ β FOR $C_n = 0$ EQUALS $-\psi$ FOR $C_n = 0$	④ C_{hr} FOR ① AND ②	⑤ PEDAL FORCE $q \times C_{hr}$	6 PEDAL FORCE IN LB	C_Y FOR ① AND ②	ANGLE OF BANK φ EQUALS $\sin^{-1} C_Y/C_L$
DATA TAKEN FROM FIGURE 12, PART (A)						FROM PART (A) FIG. 11	
0°	−9.0°	9.0° R	+.040	26.6	17 R	−.325	9.6° R
−15° R	−1.9°	1.9° R	.222	"	98 R	−.150	4.4° R
−20° R	+0.5°	0.5° L	.271	"	121 R	−.096	2.8° R
−25° R	2.9°	2.9° L	.272	"	121 R	−.040	1.2° R

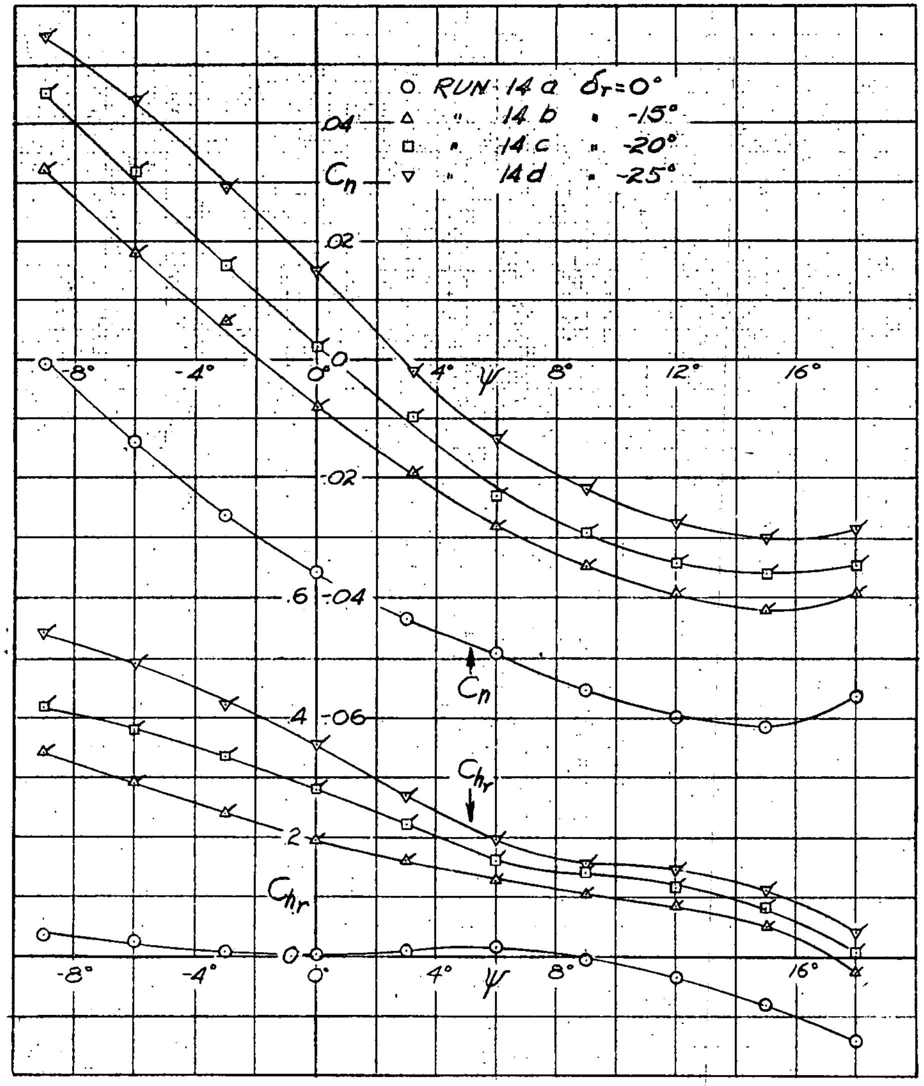
$V_i = 81$ mph, $q = 16.8$ LB/SQ FT; WING LOADING = 32.6 LB/SQ FT

(B) COMPUTATION TABLE



(C) AIRPLANE STEADY SIDESLIP CHARACTERISTICS

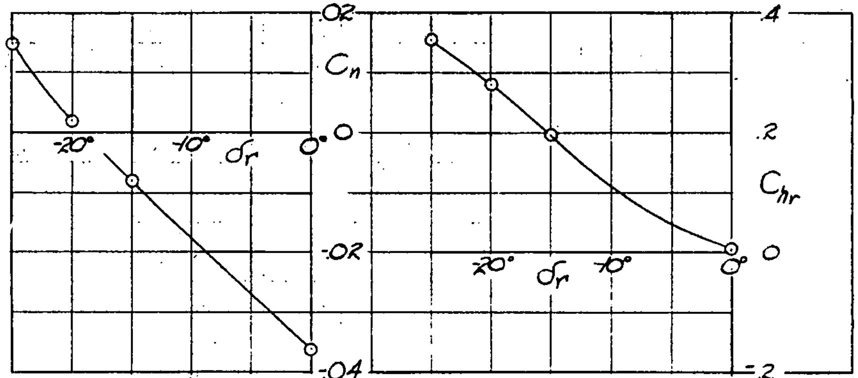
FIGURE 11.- RUDDER ANGLE AND PEDAL FORCE NECESSARY TO HOLD WINGS LEVEL IN WAVE-OFF. FLAPS AND GEAR DOWN, TAKE-OFF POWER. SINGLE-ENGINE AIRPLANE.



(A) MODEL CHARACTERISTICS DETEMINED FROM WIND-TUNNEL TESTS.

①	②	③	④	⑤	⑥	⑦
C_l FOR FULL RIGHT AILERON SEE FIG. 7 $\alpha = 6°$	C_n FOR FULL RIGHT AILERON SEE FIG. 7 $\alpha = 6°$	C_{l_p} FROM REF. 6	$pb/2V$ EQUALS $-$ ①/③	C_{n_p} FROM REF. 6	C_n DUE TO ROLLING EQUALS ④ × ⑤	C_n TO BE OVERCOME BY RUDDER = ② + ⑥
.053	-.0050	-.43	.123	-.047	-.0059	-.0109

(B) COMPUTATION TABLE – YAWING MOMENT COEFFICIENT DUE TO MAXIMUM RIGHT AILERON DEFLECTION.

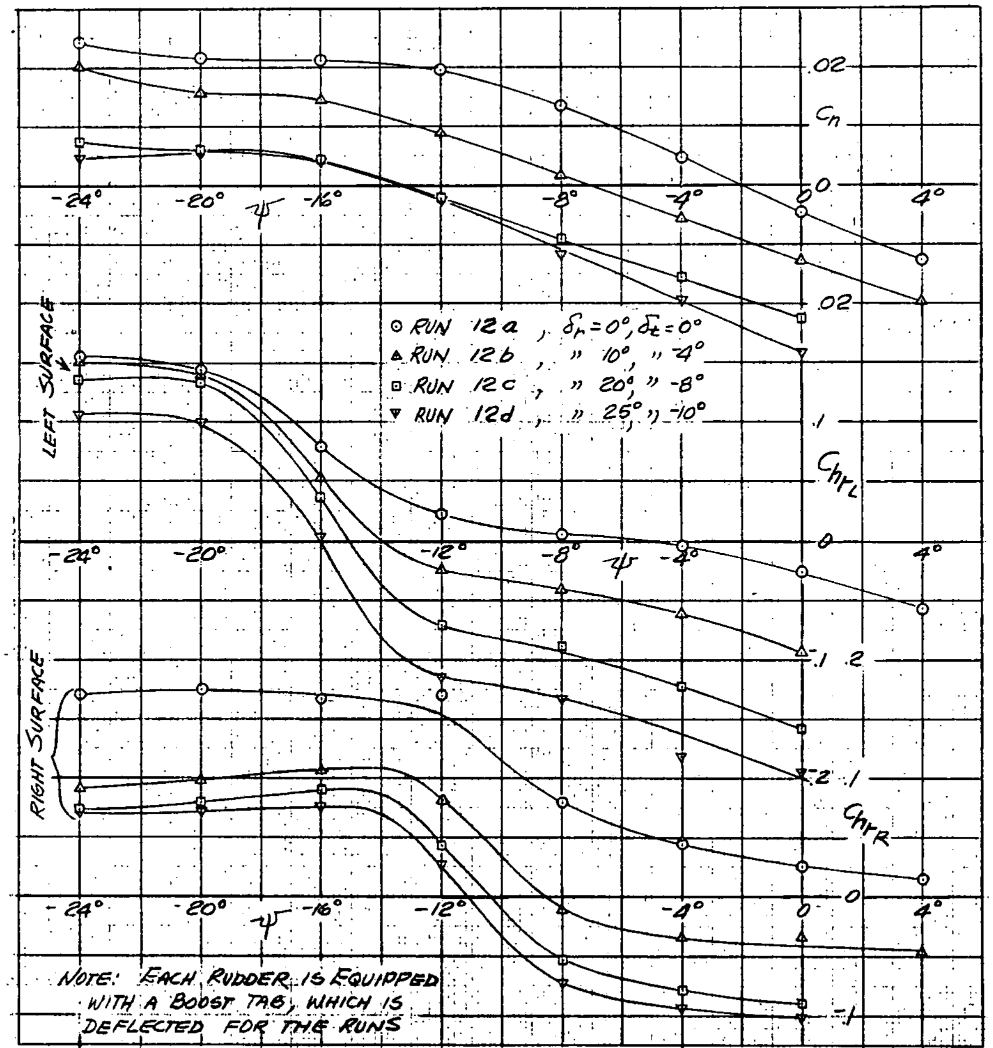


(C) MODEL CHARACTERISTICS OF PART (A) CROSS-PLOTTED AT $\psi = 0°$

①	②	③	④	⑤
C_n TO BE PRODUCED BY RUDDER. FROM ⑦ OF (B) ABOVE.	δ_r CORRESPONDING TO ①. FROM CROSSPLOT (C) ABOVE.	C_{h_r} CORRESPONDING TO ②. FROM CROSSPLOT (C) ABOVE.	PEDAL FORCE $q \times C_{h_r}$	PEDAL FORCE IN LB
.0109	-23.5° R	.335	26.6	150 R
V_i = 81 mph q = 16.8 LB/SQ. FT., WING LOADING = 32.6 LB/SQ. FT.				

(D) COMPUTATION TABLE – RUDDER ANGLE AND PEDAL FORCE ON AIRPLANE TO HOLD ZERO SIDESLIP WITH MAXIMUM RIGHT AILERON

FIGURE 12.- RUDDER ANGLE AND PEDAL FORCE NECESSARY TO HOLD ZERO SIDESLIP IN WAVE-OFF. FLAPS AND GEAR DOWN, TAKE-OFF POWER. SINGLE-ENGINE AIRPLANE.

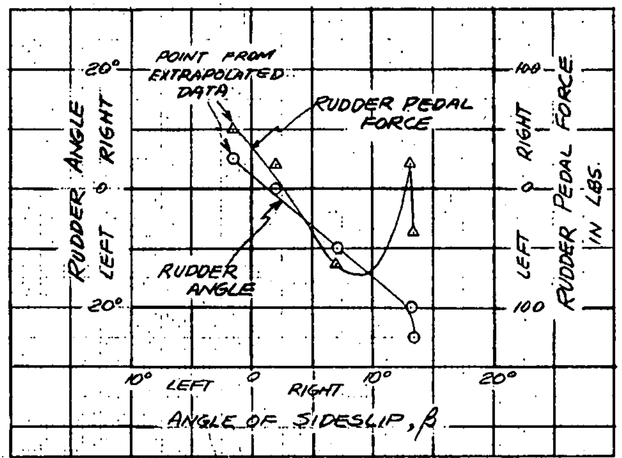


(A) MODEL CHARACTERISTICS DETERMINED FROM WIND TUNNEL TESTS

① RUDDER ANGLE δ_r	② ψ FOR $C_n = 0$	③ β FOR $C_n = 0$ EQUALS −ψ FOR $C_n = 0$	④ C_{h_r} FOR ① AND ② LEFT $C_{h_{r_L}}$	⑤ RIGHT $C_{h_{r_R}}$	⑥ C_{h_r} TOTAL, ④ + ⑤	⑦ PEDAL FORCE ∝ C_{h_r}	⑧ PEDAL FORCE IN LBS
0	-2.0°	2.0°R	-.014	+.034	+.020	38.5 →	20R
10°	-7.1°	7.1°R	-.044	-.021	-.065		64L
20°	-13.2°	13.2°R	-.055	+.075	+.020		20R
25°	-13.4°	13.4°R	-.097	.060	-.037		37L

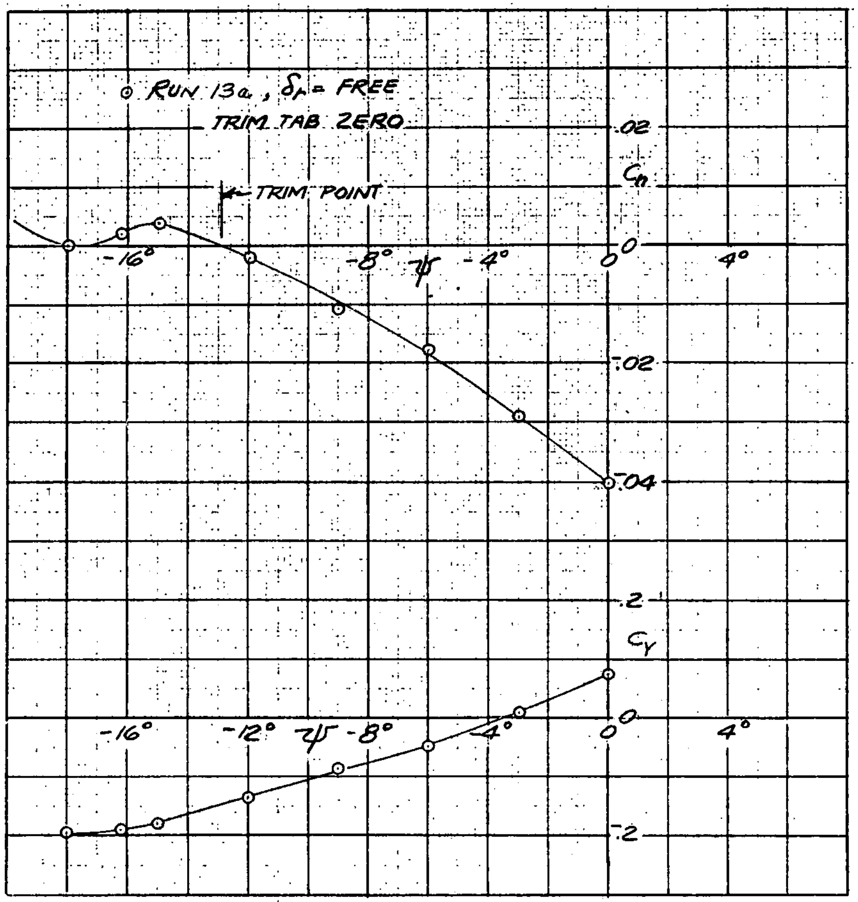
V_i = 100 mph, q = 25.6 LBS/SQ. FT., WING LOADING = 48 LBS./SQ. FT.

(B) COMPUTATION TABLE



(C) AIRPLANE STEADY SIDESLIP CHARACTERISTICS.

FIGURE 13.- VARIATION OF RUDDER ANGLE AND PEDAL FORCE WITH SIDESLIP AT APPROACH SPEED. FLAPS AND GEAR DOWN, TAKE-OFF POWER. TWIN-ENGINE, TWIN-TAIL AIRPLANE.

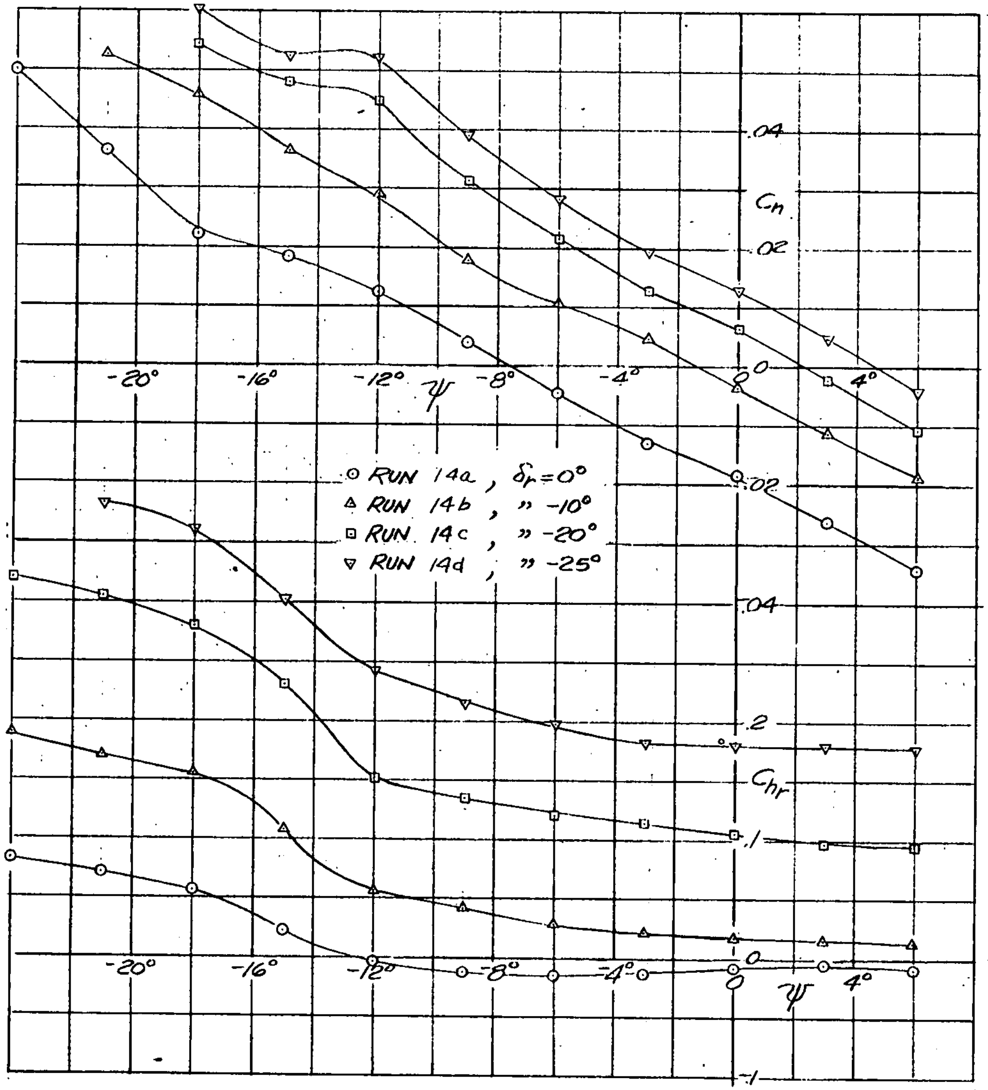


(A) MODEL CHARACTERISTICS DETERMINED FROM WIND-TUNNEL TESTS

①	②	③	④
ANGLE OF YAW AT WHICH C_n EQUALS ZERO	SIDE FORCE COEFFICIENT, C_Y, AT ①	AIRPLANE LIFT COEFFICIENT $C_L = \frac{W}{qS}$	ANGLE OF BANK, $\varphi = \sin^{-1}\frac{C_Y}{C_L}$
-12.9°	-.15	1.70	5.1° RIGHT

(B) ANGLE OF BANK COMPUTATION TABLE

FIGURE 14.- RUDDER-FREE TRIM CHARACTERISTICS WITH ASYMMETRIC POWER AT LOW SPEEDS. FLAPS AND GEAR DOWN, TAKE-OFF POWER ON RIGHT ENGINE. LEFT ENGINE PROPELLOR WINDMILLING. TWIN-ENGINE AIRPLANE.



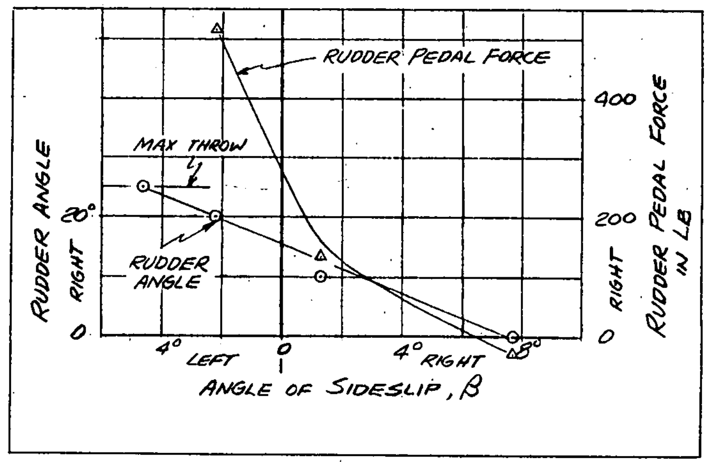
(A) MODEL CHARACTERISTICS DETERMINED FROM WIND-TUNNEL TESTS

① RUDDER ANGLE δ_r	② ψ FOR $C_n = 0$	③ β FOR $C_n = 0$ EQUALS $-\psi$ FOR $C_n = 0$	④ C_{h_r} FOR ① AND ②	⑤ ΔC_{h_r} OF TRIM TAB SET TO TRIM WITH SYMMETRIC POWER¹	⑥ C_{h_r} TOTAL ④ + ⑤	⑦ PEDAL FORCE / C_{h_r}	⑧ PEDAL FORCE IN LB
0°	−7.7°	7.7°R	−.014	+.008	−.006	1106	29L
−10°	−1.3°	1.3°R	+.020	↓	+.028	↓	133R
−20°	+2.2°	2.2°L	.100	↓	.108	↓	515R
−25°	4.6°	4.6°L	.180	↓	.188	↓	896R

$V_i = 130$ mph, $q = 43.1$ LB/SQ FT, WING LOADING = 45 LB/SQ FT

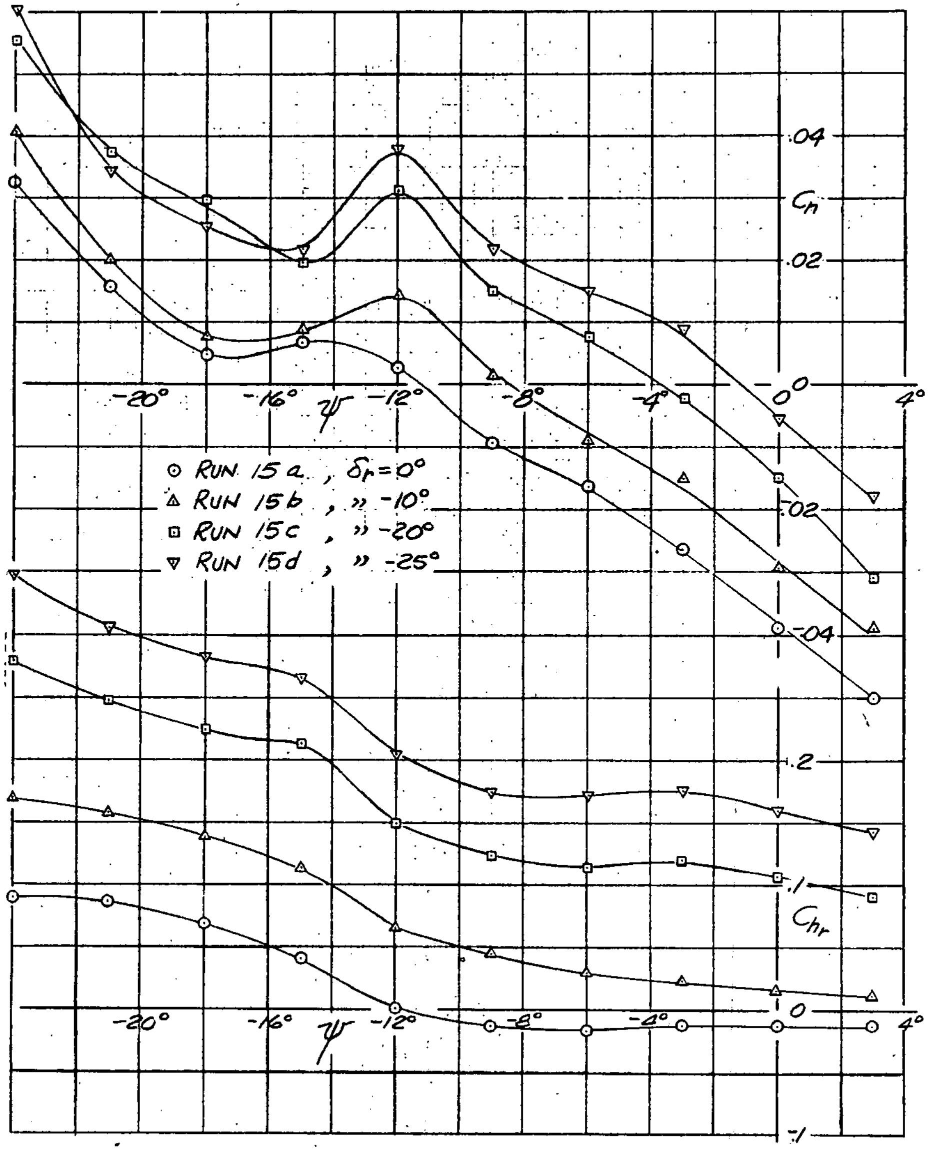
¹ ESTIMATED FROM ITEM 8a

(B) COMPUTATION TABLE



(C) AIRPLANE STEADY SIDESLIP CHARACTERISTICS

FIGURE 15.- RUDDER ANGLE AND PEDAL FORCE NECESSARY TO HOLD ZERO SIDESLIP WITH ASYMMETRIC POWER AT LOW SPEED. FLAPS AND GEAR UP, TAKE-OFF POWER ON RIGHT ENGINE, LEFT ENGINE PROPELLOR WINDMILLING. TWIN-ENGINE AIRPLANE.

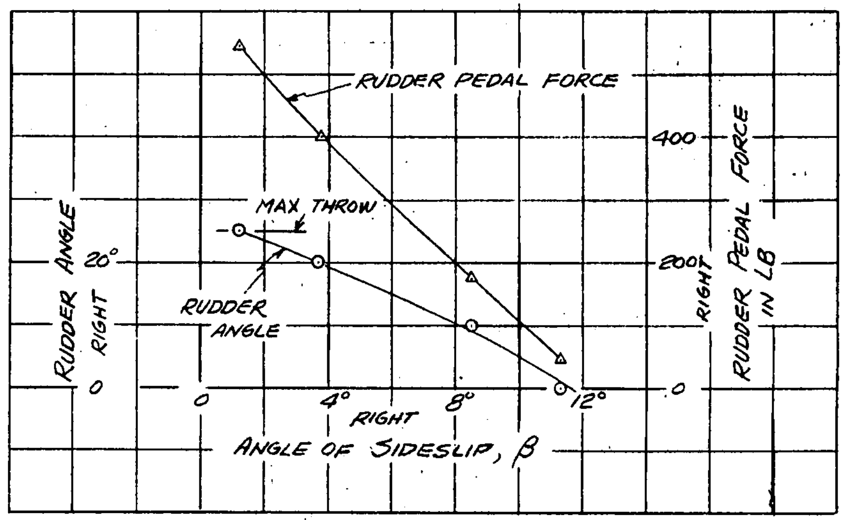


(A) MODEL CHARACTERISTICS DETERMINED FROM WIND-TUNNEL TESTS

① RUDDER ANGLE, δ_r	② ψ FOR $C_n = 0$	③ β FOR $C_n = 0$ EQUALS $-\psi$ FOR $C_n = 0$	④ C_{h_r} FOR ① AND ②	⑤ ΔC_{h_r} OF TRIM TAB SET TO TRIM WITH SYMMETRIC POWER[1]	⑥ C_{h_r} TOTAL ④ + ⑤	⑦ PEDAL FORCE / $q\ C_{h_r}$	⑧ PEDAL FORCE IN LB
0	-11.3°	11.3°R	-.003	.018	.015	110.6	44 R
-10°	-8.5°	8.5°R	+.041	↓	.059	↓	173 R
-20°	-3.7°	3.7°R	.118	↓	.136	↓	400 R
-25°	-1.2°	1.2°R	.168	↓	.186	↓	546 R

$V_i = 102$ mph, $q = 26.6$ LB/SQ.FT., WING LOADING = 45 LB/SQ.FT.
[1] ESTIMATED FROM ITEM 8a

(B) COMPUTATION TABLE



(C) AIRPLANE STEADY SIDESLIP CHARACTERISTICS.

FIGURE 16.- RUDDER ANGLE AND PEDAL FORCE NECESSARY TO HOLD 10° SIDESLIP WITH ASYMMETRIC POWER AT LOW SPEED. FLAPS AND GEAR DOWN, TAKE-OFF POWER ON RIGHT ENGINE, LEFT ENGINE PROPELLOR WINDMILLING. TWIN-ENGINE AIRPLANE.